MULTI-KILOWATT SOLAR CELL POWER

ITS CRITICAL TECHNOLOGY AND HARDWARE DEVELOPMENT

TRANSCRIPT

OF

PRESENTATION

AT

NASA / HDQTRS

13 JULY 67

BY

ASTRO ELECTRONICS DIVISION

DEFENSE ELECTRONIC PRODUCTS

RADIO CORPORATION OF AMERICA

PRINCETON, NEW JERSEY

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Part III (Cont'd)

Introduction by Arvin Smith, Chief of Solar Power and Chemical Systems, NASA-OART.

We'll start off here with my rather low key introduction while we're waiting for the other people to sign in. As we all know, we will need something that will tend to weigh less than our mechanical power systems, for our manned space missions in the future which tend to go a longer period of time. I think the recent selection of the solar cell-battery system for the Apollo telescope mount in the SIV-B workshop, on the Apollo applications program, indicates the system that we will be turning to first for longer duration electric power for our manned missions. Within OART, we are working quite hard, though, to bring along the nuclear power option. Our interest in studying the solar cell-battery system for manned space stations, as well as the nuclear system option, is to better understand how these two options relate to each other and what the advantages of one will be over the other. Today we will hear about one of the studies that I consider perhaps the most definitive study that has been done in the last 5 years in looking at the use of solar cell battery systems for the manned space vehicle. You certainly will find as we go to the higher power levels that the solar cell array gets quite large as most of you well know. In one of the configurations that you'll hear discussed today you'll find that we need 1900 sq. ft. of array -

at least that's the estimate, to produce 5 kilowatts of regulated power during the nighttime, with a little over 8 kilowatts during the satellite daytime. This is particularly important when we think about a 35 or 50 kilowatt power system. If the same ratio applies, it would mean that you would need almost a half-acre of solar cells in space for these power levels. So we are very desirous of understanding the impact of these very large areas both on the vehicle operation and the other penalties associated with it. Certainly in a low earth orbit the fuel penalties are serious if you want to stay there for a long period of time. The people from RCA will be giving their final report to NASA on a nine month study conducted principally during the latter part of 1966. Their report is out - it's a three volume report - the first volume being the summary, with substantially more detailed information for technical specialists in the various technical areas that are involved such as solar cells, batteries, and power conditioning in volume two, and further details as to what some of the assumptions were and supporting documentation in volume three. Preliminary distribution of this report has been made by the Manned Spacecraft Center in Houston who had responsibility for the technical direction of the study. Altogether, I understand perhaps 40 or 50 copies of the report have been distributed. If any of you after reflecting on it are truly desirous of having a copy of the report, if you find that you do not have a copy or that your organization does not have a copy, please write me a letter and I will see that you get a copy. If you are interested in a slightly different discussion of the subject than you will get today, one oriented more to a technical society audience, there are two papers discussing the technical results of this study that will be given this August in Miami Beach at the Inter-society Energy Conversion Engineering Conference in a workshop session on photovoltaics. One of the papers is by Mr. George Barna, who will be one of our speakers today, and we have extra copies of his paper here for those of you who are interested. In fact, I would suggest that you read through this one before you contemplate on digging into the final report. Dan Mager, who is our second speaker today, is co-author of a second paper - a companion paper that will also be given at Miami Beach. We do not have copies of his paper today but these can be made available to you if you would like to have them. Related to this overall question of solar cell power systems for manned space stations and electric power systems in general is some work that General Dynamics/Convair is doing. I have here their mid-term status report dated June '67 which discusses some of the tradeoffs of the solar cell battery system for the basic subsystem module definition study they are engaged in. In general, Houston has specified that the RCA study will form the basis for the deliberations that General Dynamics is doing in looking at the overall vehicle, but General Dynamics has included some of their own thoughts in their report. The discussion this morning is separated into three parts. George Barna, who is the Manager of Spacecraft Systems, at the RCA Astro-Electronics Division has parts one and two. After part one we will have a short break, and there will be coffee for those that desire it. Then we will come back and hear from George Barna once more and we will conclude the morning session with a presentation by Dan Mager. I'd like to introduce George Barna at this time. In terms of questions, I think we will be able to move along better if we wait until each part is completed before asking your questions. The two gentlemen from RCA have indicated a willingness to remain here this afternoon to talk to any of you who want to discuss the subject in more detail.

George Barna

Thank you very much Arvin. On behalf of RCA, I'd like to express our appreciation for this chance to talk about the study that we did for Houston. Your comment on half-acres of solar cells reminds me of a useful design constant for the folks that are really thinking big in the solar power business, that there are 2.3 acres of solar cells required per megawatt. You might keep that in mind for future use.

During this first hour what we would like to cover are the major technical details of the study that RCA did for Houston. We obviously can't cover the 800 pages of the report in the 40 minutes that we have this morning. What I'd like to do is talk perhaps for the first 5 or 10 minutes about solar power in general - what we know about it, and where the industry sits today - then take perhaps the last 30 or 35 minutes and present the details of one example that we did as part of the Houston study.

Let me then get started on the general portion of the discussion, with a statement that I think should be fairly non-controversial. Solar power has certainly been the backbone of both the NASA and Air Force space programs that have been undertaken in the last ten years. As figure 1 indicates, solar cell technology is well developed and well understood. On this figure, I have selected three examples to make a couple of important points. First of all, together they represent the very wide gamut of scientific missions that have been successfully completed using solar power. On the far left we have the planetary missions. These are significant because they represent the reliability that you can get with solar power. Planetary missions are required to remain in trans-flight for many months before using their power. In the case of Mariner IV, this period was nine months before the main experiments were turned on. Furthermore, in keeping track of Mariner IV, we've seen it come around for an additional two or three years of demonstrated power performance. There are other examples of reliable power systems in space. TIROS VII, for example, has recently completed four years in orbit at full power, taking into account the expected radiation degradation. There are other examples of long-lived TIROS satellites with three years, two years and so forth. The main point, then is that solar power has proven to be reliable.

We have the Lunar Orbiter shown as a second example. I notice that we have some of the Boeing folks here. RCA did its share towards the success of the Lunar Orbiter. However, the Orbiter is not shown because RCA had a part in it but rather to demonstrate the point that we're well along the way to at least one facet of solar array technology that will be important when we start to go into the very lightweight arrays. Lightweight arrays, particularly when used for earth orbiters, will be required to operate over very wide temperature swings. If we are talking about arrays of one-half pound per square foot, for example, in a near earth orbit, we are talking about temperature swings that will go as cold as perhaps $-150^{o}C$ up to perhaps $+60^{o}$, $+70^{o}$ or $+80^{o}C$, and it will do this every orbit. This is quite a rigorous requirement on the array. On the Orbiter program we have demonstrated that we can build an array that can go between the limits of -120^{o} and $+120^{o}C$ for perhaps a 1000 cycles and that's well along the way to solving the problems that we think we will have to face in the future.

The third program here, representing the earth orbiter missions, is important because it demonstrates both a failure and a subsequent success. One of the problems that we will have to worry about with the very large arrays is orientation in space. On the Nimbus program shown here, there was a failure that we will have to own up to in any discussion on orientation. The first Nimbus lasted only 28 days in space because of a failure in its array rotating mechanisms. The important point, however, is that there was a recovery from the failure and as of the last look the second Nimbus has performed well for over 14 months. So, we have a very important point on the curve to demonstrate that we can design reliable orientation systems.

If we take a look at a close-up of the Nimbus paddle on figure 2, we can make a couple of other points about the solar cell art. This particular panel, one of two such units used for Nimbus, measures 3' x 8' and is significant because it is an excellent building block for the very large array systems. You don't build one large array area, but rather you build many small ones and connect them together. The important point is that we know how to build these units reliably. Further, there exists a strong industrial base. We have a solar cell industry that can produce enough solar cells to supply 75,000 to 100,000 watts per year, and there are perhaps a dozen companies that can produce arrays on a large scale. We have the recent example of the ATM program to demonstrate this point.

Therefore, we are standing here and saying that we have something that we understand very well, that we have a good industrial base, and we have ample demonstrations of high reliability - so what's the big problem? Well, the big problem is size and if you will excuse my combining an astronaut with the field of oceanography - the message on figure 3 is this: if you have a large power source that has a lot of mass, a lot of inertia, and a lot of drag, you may not be able to keep control of your ship the way you would like to. This is fundamentally the problem with the very large power system. If we expand on this theme just a bit and broaden our concept of size to include area, weight, and cost, as indicated in figure 4, you come up with four fundamental problems. You have a problem with orientation, you have a problem with deployment, and you have to pay fuel penalties to compensate for the various static and dynamic perturbations that you have to contend with. But as we've indicated on the bottom of the figure, all of these problems can be handled. Let's take them one at a time.

Consideration of the impact on weight and cost doesn't let you do anything other than orient the large solar array. You simply can't afford to put three times as much array on the spacecraft as you need if you oriented the array. Just how bad is the problem of orientation? I have already commented on the success in the Nimbus program. Nevertheless, the problem of orientation seems to worry most people, all except the guys in the black hats who are worrying about Brayton and Rankine cycle systems - they seem to like to rotate things very fast. Well, in addition to the Nimbus experience, we have another good point on the curve from the OSO program. OSO-1, for example, has demonstrated 16 months of failure free performance. In addition, there are a couple of Air Force study programs - Lockheed has done one, and Westinghouse has done another, with significant results. I think the general feeling of the industry is that you can design reliable orientation hardware. Our own posture in the study program that we've done for Houston is this - yes, we recognize it as a problem, but we've seen demonstrations that the problem can be overcome. So, we'll tag it as a major problem, and we'll tag it as an area that we want to do a lot of development work in and we'll want to demonstrate the reliability by special test programs but fundamentally orientation is nothing to shy away from.

Alright, let's talk about the deployment problem. We certainly have some very fine examples of deployment in space. In figure 6 are the two biggest examples of systems deployed in space, but there are many others - the Nimbus program, the Lunar Orbiter program, the Ranger program, the Mariner program - the arrays in all of these programs have been deployed, quite simply this is true, but neveretheless they represent significant experience. Figure 6 shows the two biggest examples. Pegasus, when extended, measures 14' x 96'. Referring to the Agena, I don't really know how many of these have flown, and I can't honestly tell you there have been no failures in the deployment mechanism because the programs are classified. But to the best of my knowledge they have worked reliably. Furthermore, they are going to be used on the SERT II program and also for some future missions in the manned space program. So, our posture on the study has been that as long as we use the simple deployment mechanisms that have been successfully demonstrated, we can handle the deployment problem.

Well, let's then talk about the latter two problems in figure 4 - they are coupled together - the problems of static and dynamic interaction and the associated fuel penalties. In the area of static perturbations first - here we're talking about the natural forces that you're going to have to contend with in orbit. These are drag forces, solar pressure forces, gravity gradient torques and magnetic dipole torques. We have sized the problem in terms of its impact on fuel penalties for the example that I am going to give you in a few minutes. It is a significant problem from the standpoint of fuel penalty, but it is not an overpowering problem. When you see some of the numbers for the fuel penalties, you will agree that static perturbations are not an overpowering problem.

The dynamic problem is a little bit more complex. Let me define quickly what I mean by the dynamic problem. You can picture that you have two systems with individual personalities coupled together mechanically - an array that wants to look at the sun and a spacecraft that wants to look down at the earth. Each has its individual servo system trying to make it operate in its fundamental mode. They are coupled together mechanically, with a large realm of possibilities relating to the nature of the mechanical coupling. The problem here is the obvious one. When the array tries to orient itself to look at the sun, it's going to have a reaction on the spacecraft which will try to correct for the perturbation forces that the array has introduced. The reverse is also true: when the spacecraft moves to orient itself to the earth, it will tend to disorient the array and couple perturbations into the array servo loop. This in effect makes one big overall servo loop that you have to contend with, and it has a large impact both from the standpoint of fuel penalties, which we don't feel is the major problem, and in the overall system stability, which we feel is probably the major technical problem that has to be looked at when you look at this entire technical field. This will be a problem that Dan will take quite a bit of time to look at during the third part of the presentation this morning. That's enough discussion of the general kinds of things. Let's talk about a specific example. We did for Houston a general purpose study based on fairly general ground rules for both the mechanical and electrical requirements, to which we configured an overall power system. Figure 6 shows the major constraints that we had to consider during the course of the program of which the first three are the most important. First of all, we had to look at the adaptability of the power system design to missions that would range from 200 nautical miles to synchronous altitudes, with inclinations that range from 30^{o} to polar. Rather than look at the complete spectrum, we looked at three specific examples:

200 miles at 30° and polar, and synchronous at 30°. The second major constraint we had to consider was a one year reliability goal of .995. We can stand here and tell you about the performance of two, three and four years that have been achieved with some spacecraft, but when you look at the actual cases they were not designed to meet these kinds of reliability factors, and when they were launched they had much lower predicted reliability figures. When you have to design to a goal of .995 this becomes a fairly difficult design problem.

The third major constraint we had to consider was the spacecraft attitude, which was required to fly belly-down. Therefore, whatever requirements we came up with for orienting the array, we had to contend with the fact that the spacecraft would not be much help in orienting the total system. We compensated for this by making the decision to orient the array in two axes with respect to the spacecraft. The main axis of orientation requires continuous tracking during each orbit. The second axis has a period of one year, and requires very gradual orientation.

The other two factors on figure 6 simply summarize the other constraints that we had to live with. Within the constraints of the LEM shroud, we had to integrate the power system to a Houston designed MMSS, which is basically a large tuna-fish can 15 ft. in diameter and 8 ft. high. We had to pack our system in the volume between the MMSS and LEM shroud.

We approached the study actually in three steps, as shown in figure 7. I would like to show you briefly what we did. First, we had to design a system to meet the basic requirements. Our procedure was to take the basic power and orbital requirements, and perform an energy balance study and size the power system components. The problem with this system is that it doesn't have the kind of reliability that we need, of 0.995. Therefore, we must enhance the system to bring it up to the 0.995 reliability status as the second step in the process. Now, having defined fundamentally what we can call an electrical system, the third part of the program is to look at all the perturbation forces, to look at all the mechanical considerations and finally come up with the total system configuration, which then determines the total system weight.

What I would like to do this morning is walk you through this process, with one single example. However, before you can actually get into tradeoff studies and the actual system design, you must make some fundamental decisions, relating to the problem of array reliability, battery reliability and the nature of the electronics system. Figure 8 shows that in the area of the electronics system, there are really four ways you can go. There are many, many, variations and if you look at all the space programs and the nature of their power systems, there are probably 30 or 40 different variations. But fundamentally, they can all be summarized into the four categories, of being either series or parallel, and trackers or non-trackers. Let me define my terms. A tracking system is a system where you design a little bit of intelligence into your regulation loop such that you force the power source, in this case the solar array, to operate at its maximum power point and transfer maximum power for as long as the loads can support it. In effect the system does not constrain the bus voltage, but allows it to assume whatever voltage is required to transfer maximum power. A non-tracking system on the other hand doesn't track the maximum power point, but operates over a narrow range of operating voltage regardless of the condition of the array. In terms of series and parallel systems, the main functional element is in series between the source and the load for the series types, while for the parallel types the main functional element is in shunt or parallel with the load. In the top example in figure 8 we have regulators in series between the source and load, and in the parallel system in the third example they are in parallel with the loads. In the tracker system, the series tracker is between the source and load and in the parallel system they are essentially in shunt, in this case feeding the batteries. In order to select the optimum system we went through both subjective and objective reasoning to select the system to subject to further analysis.

We first looked at the relative array efficiency. A rating of 1.0 is best and a comparison of two numbers gives you a relative idea of the different sizes of array required to support each system for the same load requirements. For example, a rating of 1.0 is 15% better than a rating of 0.85 in terms of array size. We then looked at each system in terms of baseline reliability. Remembering back to the scheme that we are developing, each system will have an inherent baseline reliability from which you build up to the required reliability of 0.995. The reliability numbers shown are those for the baseline systems, each capable of supplying the mission loads. The third column indicates what it takes to get from the baseline reliability to the 0.995 required reliability in terms of the total number of power modules. There are other modules, such as control modules, that are needed to complete the system, but the power modules are the numbers that are important. Next we looked at subjective kinds of things, for example, system dissipation. What thermal loads do you pump back into spacecraft, when using each of the systems? Other considerations were bus voltage excursion, and finally how much redesign was necessary to accommodate changes in mission requirements. Now quite quickly, we eliminated the non-tracking systems, because for the wide range of mission requirements that we were trying to meet, we felt that we needed the flexibility offered by a tracking system. Figure 8 shows that when you get to choosing between a series and parallel tracking system, you almost have to pay your money and take your choice. Primarily on the basis of the relative system efficiencies, we selected the parallel tracker for further study as the baseline configuration.

Let's take a look and put some numbers in and see how the system comes out. Figure 9 summarizes the results of this. Again a reminder, we are talking about a baseline system that will have a reliability of about 42% for one year's operation. Looking at the details, first we put into the computer program the load requirement, 8.2 kilowatts for daytime and 5 kilowatts for nighttime. Roughly 1/3 of the load requires AC power, while the other 2/3 requires regulated DC power. For the moment, the factor identified as resupply is not significant. I will talk later about resupply. The mission altitude is 200 n.mi. and the radiation flux used in the analysis corresponds to that altitude.

We have had to make some judgments based on a variety of tradeoffs by this time which I won't go into in any detail today, such as selection of the solar cell type, and specification of the solar array weight density. This latter area is a subject all its own. Based on the tradeoffs between the thermal parameters and the mechanical load parameters, we selected an array weight density of 0.81 pounds per square foot. This design allowed the array to sustain both the launch loads and the loads imposed by the SPS engine firing and docking loads, although even with the selected system, we have to orient the panels in a favored direction, to operate in a tension-compression loading mode, to withstand the SPS engine firing load. Thus, the system can stand all the mechanical loads we would expect to see in orbit, without having to retract.

Now, having defined our input and our mission requirements, and having already selected the form of the system, figure 9 summarizes most of the information that comes out from the system calculation. The most important figure here is the total system weight. For the specified mission, with a 42% reliability for one year, we come up with a system weight of 3862 pounds. This includes batteries at 1980 pounds, electronics at 489 pounds, and it turns out we need 1720 square feet of solar array to support the mission. Now that much array gives you over 23 kilowatts when it comes out of darkness, which points up the problem of thermal dissipation. On the average, the array produces 15.3 kilowatts. Based on .81 pounds per square foot, the array weighs 1393 pounds.

We now want to take a look at the next step of the analysis. We want to take the baseline system and we want to improve its reliability to .995. Looking at figure 10, the top line shows our basic system, with 42% reliability. The difference between the weight shown on this line and the weight shown on the previous chart is the weight that we have added to account for the boom and the orientation system which are not included in the computer program numbers. The two numbers represent the same basic system. The 1720 square feet is shown here - the battery type is nickel-cadmium and the numbers of electronics modules are shown on the right. Based on the choices we made in the sizing of the individual components, the 30 battery modules represent 5 parallel rows of batteries, 30 cells high, configured in modules of 5 cells apiece. Now in order to improve this system to a system which has a reliability of .995 with no resupply - in other words without requiring the system to be supplemented beyond the initial launching, it turns out that it is necessary to supplement the basic system with an additional 714 pounds of electronics and batteries, as shown by the second line. We do not have to enhance the array because we have assumed that by taking into account the various degradation factors and multiplying them linearly rather handling them in any sort of r.m.s. fashion the array itself will be adequate for the one year mission.

In the area of the battery, it is difficult to come up with meaningful, hard reliability numbers for the batteries. For our study program, we studied all the data that was available from Crane, from our own tests, and from other industrial sources.

Based on these data we postulated a cycle life performance curve, with cycle life as a function of depth of discharge. By analyzing the individual data we generated a gaussian distribution curve centered about that performance curve. We then drew a second line parallel to the original curve through the minus 3 sigma performance point on the series of distribution curves. I cannot reproduce all the details of reliability analysis but the specialists tell me that this approach results in the predicted reliability for a single battery cell of 0.9987, and you will have to take my word for it. When you take the number for the individual cell reliability and work it into a total system, we get overall performance figures that are compatible with our system needs. The individual reliability numbers for the electronic units are derived on the basis of circuit effectiveness and parts count, based on an evaluation of the parts that we anticipated would be utilized for the circuits. The numbers on the right summarize the details of the required reliability enhancement. In the first case you have to add two units, for the inverters you have to add five units and in the third case you have to add three units. In the battery area three extra modules are required, equivalent to half a string, to supplement the basic battery complement to achieve the required reliability.

There are two other system options that can be looked at.

Let's postulate that you do not quite have the necessary launch capability, and we have to look for other ways to get the system up and still maintain the basic reliability. The obvious route is resupply. If we are just a little bit overweight we can launch a system that weighs under 4400 pounds, which will be supplemented with an additional 360 pounds 45 days later to enhance the system to the reliability that we need. In this system you end up with the same total weight. It is fundamentally the same result - we have just solved it a different way.

Now let's postulate that we are really strapped for weight and have to do something fairly drastic. Let's review the selection criteria for nickel-cadmium vs. silver cadmium storage batteries. The analysis that we did indicated that up to around 2,000 n.mi. altitude, the nickel-cadmium battery is a better bet than silver-cadmium on the basis of cycle life considerations. But that doesn't rule out the use of silver-cadmium cells for these altitudes. Therefore, we can postulate the use of silver-cadmium cells for a mission of 200 n.miles, resulting in a system of about 1000 pounds rather than the roughly 2000 pounds we talked about for nickel-cadmium cells. The problem is that from the standpoint of the cycle data that we were able to accumulate, you have to resupply the silver-cadmium batteries three times during the course of a one year mission. Which says that we have a lot of weight to put up into orbit, and we have to put the weight up in three launches. The advantage is that you only have to launch 3100 pounds, regardless of the total weight penalty paid for achieving the system reliability. The number of 18 battery modules for this option implies a lesser number of battery modules, but it is based on the fact that the basic silver-cadmium cell that we picked was 200 ampere hours compared to approximately 120 ampere hours that we selected for the nickel-cadmium system.

These then are three approaches that can be taken to come up with a system which has the reliability that you require. Now that we have a system with the required reliability, let's look at some of the mechanical problems. One of the important mechanical design choices is the shape factor of the array. We looked at a large number of shape factors, and finally homed in on the two shown in figure 11 as being the most important shape factors to consider. The Z shape is attractive because it gives less of a shadowing problem. However, the perturbations problems can be quite severe. The converse is true for the H shape factor, which is arranged close to the spacecraft. This can create a shadowing problem, but from the standpoint of perturbations you are much better off. Figure 12 summarizes some analyses of the perturbation problem, which we have outlined from the standpoint of one Z configuration and two different aspect ratios of the H configuration. When you look into the perturbation forces that you have to put up with, these are the numbers that result. The predominant force of course is drag. For these mission constraints, it will cost about 1800 pounds of fuel in the course of 1 year. If you look at the lift forces, resulting from appropriate solar array attitudes that can cause lift, they can cost another 460 pounds of fuel. If you look at the gravity gradient torques, in the case of the Z configuration you have to worry about gravity gradient in both the pitch and roll axes, and again you have some fairly large fuel penalty numbers. For the H configuration you have to worry about gravity gradient only along the pitch axis - you don't have to worry about it along the roll axis because you have essentially a balanced configuration about that axis. The other perturbations - solar pressure and magnetic dipole - are quite small and insignificant when compared to the drag, lift, and gravity gradient forces. When you analyze the nature of the variation of the lift and gravity gradient forces, it turns out that both forces are cyclic in nature at either the orbital frequency or at a multiple of the orbital frequency. This lends itself to the use of momentum storage, for example, to balance out the cyclic variations. And in fact the second set of numbers that you see in figure 12 indicate just that. With momentum storage we essentially get around the problems of compensating for the lift perturbations and the pitch axis gravity gradient perturbations. It cannot, however, compensate for the roll axis gravity gradient forces. The momentum storage system will weigh less than 100 pounds, including a 35 pound reaction wheel, some electronics and a little bit of power to power the system. So we are talking about a pretty good weight tradeoff. Looking at the results then, they tell us that we do not want to go the route of the Z configuration but rather we want to select one of the two H configurations. When you look at the two choices, there is no difference when you look at the analysis in figure 13, when momentum storage is used in both cases. Therefore, the choice must be made on the basis of other mechanical design parameters. When we start to worry about the SPS firing, docking load, and that sort of thing, it turns out that the lower aspect ratio gives us a lighter weight design, and hence was our choice. So that's our perturbation story. Figure 13 shows what the system looks like when you put it all together. I haven't commented on the form factors of the system packaging, but we did look at two different ways to do it. You can postulate an approach where if there are one of two basic manned mission modules, you can store all of your electronics and batteries inside the modules and package the array between the manned modules and the shroud. This gives you many problems with interfaces. We ended up favoring something that we called the power system module or the PSM. This approach gives you a completely integrated power system built around an air-lock which is needed as part of the system. The advantages are that you end up with very simple interfaces between the power system and the basic module. You also have a good, and convenient, growth capability allowing us to virtually get twice as much power system within the volume specified to us. So we ended up favoring this approach. This means some increase in the total program cost - you now have a complete structure design to generate and you have to design a component housing and worry about such things as thermal control which perhaps you wouldn't worry about if the power system were part of an overall integrated module. We have a model here which you can examine and we can talk about during the coffee break.

It still remains to look at the shadowing problem to see how severe it is. Figure 14 shows the module at three sun angles and it gives you some perspective on how the vehicle would shadow the solar array. We analyzed the shadowing problems just this way and these are photographs of a small model that we set up on a table with a lamp to simulate the sun. We moved the lamp around while we took photographs, and we analyzed the amount of shadowed area by counting the squares that you can see on the array. By counting squares at the various angular configurations, you can generate a performance curve of sun angle vs. the amount of shadowing on the array. When you do this you come up with the curve on top in figure 15, which says that as I vary the sun angle from 0 to 90° the amount of shadow loss that I take and the effective amount of array that I have illuminated varies in this manner. However, there is a compensating set of conditions that you have going for you. It turns out that over the same variation of sun angle the amount of sun time per orbit is increasing, so that for the 200 nautical mile orbit that we looked at, the two conditions tend to compensate for each other and you get the curve on the bottom, which essentially says that for most of the sun angles that you are looking at the power from the system is virtually constant. It does tend to go up just a little bit, then tends to fall off over the last 10 degrees, for sun angles greater than 80°. There are a number of ways that you can compensate for this falloff, such as just turning off some of the experiments during that period. Another solution is to roll the vehicle. Fundamentally it is not as large a problem for this mission as we had thought that it might be, and have not compensated for it in terms of a larger array.

Let's summarize. Figure 16 summarizes the example that we have just gone through. Starting with our solar array, 1720 square feet, these are the various electronics components that we talked about. For example, here is the charger tracker that forces the array to operate at its maximum power point as long as the load can accept the power. In each component block we have indicated the number of baseline units and the number of spares that are required for one year's operation.

In the upper right corner is the power profile. To review, we are talking about 8.2 kilowatts during the daytime and 5 kilowatts during the nighttime. In the lower left corner, we show what the system will look like during launch. The array sections can be seen in their folded position. The orbital configuration is shown in the other view. The total system weight is also summarized. We've included the total fuel penalty and as part of the system weight, based on using momentum storage, The total system weight is 6600 pounds, where I have indicated that this is less the structure weight. We left the structure weight out because the airlock is not defined at this point. If you want a total picture, a good estimate for the structure weight would be about 15% of the power system weight. To realistically evaluate the power system, you shouldn't penalize it for the structure weight to get its total performance.

We have broken the weight numbers down, to understand what we have, which I've tried to show in figure 17. If you take the numbers in figure 16, for example, the array, which we said was some 1500 pounds and could produce 15 kilowatts on the average, the array weighs about 100 pounds per kilowatt. Similarly, if you go through all the other components, the battery works out to be 390 pounds per kilowatt, and so forth as shown in figure 17. For the batteries, this is at first glance an odd kind of number, since we normally think of batteries in terms of pounds per kilowatt-hour or ampere-hour, but for our purposes here this number is worked out in terms of kilowatts rather than kilowatt-hours, based on 5 kilowatt nighttime load. For the charger tracker and power conditioning, here the numbers are based on the weight of the basic units in addition to the units that are required to enhance the reliability. So that even though the figure of 90 pounds per kilowatt may seem on the high side to you, they are equivalent to a per unit weight on the order of 25 to 30 pounds per kilowatt which is more or less consistent with current practice. We have also taken the total fuel penalty and prorated it on the basis of the total array capability of 15 kilowatts. Now let's take the system and let's break it into two parts, into one part that we term daytime loads and a second that we call nighttime loads. We will break out the power conditioning separately. For 8.2 kilowatts of daytime load at the regulated bus the system will effectively weigh the sum of the daytime and power conditioning blocks. For the 5 kilowatts at nighttime the system will weigh the sum of the nighttime and power conditioning blocks. The X's in the upper tables identify which components relate to the daytime loads and to the nighttime loads. It turns out that when you look at the system this way the daytime system to provide 8.2 kilowatts works out to be about 2.8 watts per pound; the nighttime on the other hand, works out to be about 1.1 watts per pound. The total system is also close to 1.1 watts per pound. Actually, the nighttime number alone is a shade over 1.1 watts per pound. If we look at the total system weight without the fuel, the figure increases to 1.5 watts per pound. Comparing this to a reactor-thermoelectric system which is one of the competing systems for this power range, the kinds of performance figures that were presented at Lewis last Fall were 1000 to 1500 pounds per kilowatt for a man rated reactor-thermoelectric system. So our system is in the ball park even when we include the fuel penalties.

During our study we really tried to stick to things that are consistent with the sorts of things we can do today. If we go one step further there are perhaps one or two improvements that you might look for in the not too distant future. We can take a look at the array, and postulate that we can certainly look forward to perhaps 50 pounds per kilowatt - remembering that when you are looking at 50 pounds per kilowatt or 20 watts per pound, we may not be able to make it work for this mission because of thermal or strength parameters but let's assume that we can make it work for our mission. Let's also assume that we can take silver-cadmium batteries and improve their cycle life so that they become competitive at the low altitudes that we looked at. When you go through the same analysis as in figure 17, the analysis summarized in figure 18 results. The table here is exactly the same as in the previous chart. When you go through the numbers, there is a modest improvement in the daytime system, to 3.4 watt per pound. For nighttime there is a fairly significant improvement to 1.7 watts per pound, and with or without fuel the system performance is becoming quite attractive. This completes about all I can run through in the available time, although it seems I'm five minutes ahead of time. I guess the best thing to do is ask for questions at this point.

George Barna - Part II

During the first hour we have talked about the technical details of what a typical large manned mission solar power system might be. As figure 19 indicates, what we would like to talk about for the next 20 minutes to one-half hour, is what we might look for in terms of the total program schedule, and the total program cost.

Let's look first at the overall program schedule shown in figure 20. As part of the study, we looked at what it would take to go from the system concepts that we have talked about, through design, to a flight acceptance tested piece of hardware. When we put all the tasks together it comprises a 39 month program. We have divided the program into four phases, consistent with the way we would develop such a program at RCA. Basically the first phase is fundamentally the paper design phase, and is 9 months long. The output from this phase will be a set of detailed drawings ready for first piece fabrication release, a set of detailed performance and procurement specifications; test plans, test procedures, test specifications; and designs for the various pieces of handling and test equipment which are a very significant portion of a program like this. During this phase, we would also expect the circuit design engineers to breadboard their circuits to prove that the circuits will do what they were designed to do. In the area of batteries, since we're really designing around basically a new product and we don't have a great deal experience with batteries of this size, we would contract the battery manufacturers to design the cell and the first run of these cells should be put through characterizations tests. In the area of the solar array development, we mentioned that we were quite concerned about the very large temperature swings on the array that may go as cold as -150 and up to +70°C. During phase I, we would build sections of the array and put them through extended cycling programs to verify that the solar cell interconnections and method of bonding the cells to the array substrate, and in fact the substrate itself, are compatible with those temperature extremes. From this phase then we gain a good deal of knowledge and confidence, prior to committing the design for hardware fabrication. Going to the other end of the program spectrum the prototype and flight hardware are basically one and the same, with no real difference except that the prototype will be stressed to perhaps one and a half times the levels to be seen by the flight hardware. But basically they are handled and built the same way, and are not really hardware that the engineer can turn knobs on. The engineering test model therefore is the hardware that bridges the design phase and what we might call the formal hardware phase, and is in fact really a learning phase, where the engineer will be able to turn knobs and make changes, before finally making a release for prototype and flight hardware. I'd like to point out one thing on the schedule. It looks like we have a great deal of overlap built into the schedule, but it is really not as bad as it looks. The overlap is intended to show that there are some fairly long lead procurement and parts accumulation cycles. We are talking about many, many piece parts which will take a long time to accumulate. However, the schedule is based on not beginning fabrication of the prototype before the testing of the engineering test model is fabricated, had been fairly well wrung out in test programs and had been design reviewed. The prototype release point occurs somewhere around the 20th or 21st month in the case of the solar array. The same constraints exist between the prototype and flight units, where we waited until the bulk of the prototype testing was completed before we initiated the fabrication of the flight model hardware.

Let's look at two of the major component problems that we have to contend with in terms of lead times. I should have mentioned during the first talk that one of the tradeoffs that we looked at was which kind of solar cell should we use on the program? We looked at tradeoffs between 2 centimeter x 2 centimeter, 2 centimeter x 3 centimeters, and 3 centimeter x 3 centimeters solar cells. We looked at whether the solar cells should be 15 mils thick, 10 mils thick, or 8 mils thick. We looked at whether the cells should use wrap-around contacts or conventional contacts. From the various tradeoffs which we looked at from the standpoint of program costs, from the standpoint of system weight, from the standpoint of packing factor, we ended up with the recommendation that the cell that we would recommend would be the 2 cm x 3 cm solar cell, 10 mils thick, with conventional contacts. The number on the chart relates to the total number of 2 cm x 3 cm solar cells that are required, totalling 320,000 solar cells that we would have to procure for the entire program. In terms of equivalent quantities of 2 cm x 2 cm cells with which most people are calibrated, this quantity is the equivalent of about 500,000 2 cm x 2 cm solar cells. So we're talking roughly one year's time to accumulate approximately 500,000 equivalent 2 cm x 2 cm solar cells. If you break this down - and let me stay with the 2 x 2 cm cell equivalent for looking at the delivery rate - assuming a lead time to first piece delivery of 4 to 6 weeks and some appropriate build-up rate - we are talking about delivery of solar cells at the rate of 12,000 to 15,000 solar cells per week. This is a healthy rate of delivery. Nevertheless, as of today, I can tell you it can be done because it is being done for the ATM program.

If we talk about panel delivery rates, the numbers are perhaps not quite as striking as the solar cell delivery rate, but remember that we are talking about producing about 5,000 square feet of array for the three systems. We assumed that two sections of the total array would be adequate for the engineering test model, and for the prototype and flight systems we would certainly build the total area of the array system, as all four sections. So we are talking roughly about 5,000 square feet of panel area. When you break this figure down, this means approximately 100 square feet of solar array per week. Since we recommended basic building blocks of about 6 square feet we are thus talking about delivering

16 modules a week. Considering that the delivery rate of panels is also the rate at which we have to fabricate live solar panel area, this says that every week we are going to deliver the equivalent array area of two Nimbus spacecraft, where each Nimbus is about 50 square feet. This then sizes some of the major delivery and manufacturing problems. Looking at even this level of technical and schedule detail, it is fairly obvious that a large amount of test equipment is required to support a program of this size.

Let's look at the cost for the total program, shown in figure 21. The total cost for the program we have estimated at 33.8 million dollars, with the costs distributed into the four program phases as shown. Arvin will argue with the total cost reflected here - he thinks that it is too high and perhaps he's right. There is some indication that at least in the area of the solar array, we can make some savings and perhaps get the number down to 30 million. Let me make one other point. We are talking here of the costs of the total PSM system. This means that in addition to the fundamental power components we are going to build a very large structure, including a component housing with its thermal controller. It also means we are talking about integrating all of the components into a total system and testing the total unit as an integrated power system. The cost figure that I would like to call your attention to is the one for phase 4, because this is the repeat cost for one flight system, 9.35 million dollars. If we generate a figure for the average power capability for the power system, we come up with 7 kilowatts, based on a capability of providing 8.2 kilowatts during the daytime and 5 kilowatts during the nighttime. If you are interested in a figure of merit for the cost, it turns out to be about $1300 a watt for repeat orders of the power system after the engineering design is paid for. That price would be the same for the second, the third, and the fourth unit. However, if you ordered 100 systems, we could probably work out a nice price break.

Let's look at the cost a little differently. Let's first identify the major system elements from the design point of view, shown in figure 22. There are four components that feed into this thing that we have called the system; the electronics, the battery, and the array which we have broken into two component groups. There is the array proper, which is the active array area which interfaces with the boom. Then there is the deployment and orientation system which includes the boom, all the rotating components and the electronic servo which is required to orient the array system.

All of the components fit into the large structure, which includes a thermally controlled compartment for the electronics and batteries. Therefore, we have to design a structure around an airlock that would be furnished; also the component housing, and finally, harnessing, patchboards and other common elements. Then it is required to test this completely integrated system. This is quite a sizable task.

If we look at the cost that pertains to a breakdown by a major system elements, the costs break down as shown in figure 23. I've added a sixth element to account for the cost of the Project Management Office. The single biggest cost item is obviously the array, which is half the total cost of the program. If I start with the smallest item, the PMO costs account for 5% of the total program cost. This is a reasonable number based on past practice. To explain the battery costs, let me indicate roughly what has to be produced. We need three complete systems complements of batteries. If we are talking about 33 battery modules per system, and in addition we have to include some spares, we are talking about a total number of battery modules to be produced of over 100 units. The costs shown in the chart reflect this level of production.

In the area of the deployment and orientation costs, we have a couple of major technical problems that require some rather elegant test programs. For example, we have to demonstrate reliability of the slip rings, and the various rotating components. We propose that special test programs be run to evaluate the rotating components for long periods of time in vacuum to demonstrate their reliability.

In the area of electronics, the total cost breaks out as roughly a million dollars per phase, with an additional quarter-of-a-million dollars in each of the second and third phases for test equipment and facility costs. We must develop ten unique designs of electronic modules, which will result in a total of about 30 black boxes per system, with a complete complement required for each of the three program phases. The total electronics complement will weigh about 1,000 pounds. Therefore, altogether we are talking about approximately 3,000 pounds of electronics. This works out to about 1,000 dollars per pound of electronics if that figure means anything to anybody.

Coming back to the system again, the costs include the basic structure design, the component housing design, and it includes looking at the system design particularly in relation to its dynamic interaction with the spacecraft. It includes integrating the components into the system, and testing the entire assembly then as a complete system. If you're interested only in a power supply cost, perhaps only a million dollars of the 6.2 million dollars noted here is really associated as power supply cost. The remaining dollars result from the integrated system approach. Therefore, from the standpoint of a pure power supply cost, the total program cost is perhaps 5 million dollars or so less than the 33.8 million dollars that we previously reported.

Let's now look at the array cost. Let's first identify what I'm talking about when I talk about the solar array. Figure 24 shows the solar array as I've defined it. During the launch mode it is all folded together as shown in the top view. The center view shows the array as it unfolds. The bottom view shows the unfolded assembly. There are four sections like the one shown in the figure, to comprise a total array system. The dimensions for an unfolded section run approximately 17 x 28 feet. Each section is made up of 10 panels which are in turn made up of 8 modules, each about 6 square feet. Therefore, we are talking about panels that are approximately 48 square feet, and each section is about 480 square feet. Let me now define what we recommended for each of the three hardware phases. For the engineering model you don't have to build the complete system of four sections. We felt that we could derive adequate engineering information of the type we needed with only two sections. At the same time you don't have to have array panels, complete with solar cells. We felt that dummy panels using glass or aluminum chips to simulate the solar cell would be adequate for engineering tests. You do have to build one or two live panels to characterize the compliance and some of the other important characteristics of the panel, but other than that, and a desire to vibrate a live panel, you can effectively use dummy components. Similarly, for the prototype we think you should build one completely live section, but we felt that the other three sections could also use dummy components. The flight model obviously has to be four live sections. Let's look at the costs, shown in figure 25. Starting at the flight model end of the program, we need to produce about 1900 square feet of array, to which must be added some process rework requirements. We assumed a unit cost of $2800 per square foot for building the array. Let me identify what that number means. It pays for all the materials such as solar cells, glass, adhesives, and the interconnection strip, and for the complete fabrication process to produce finished array sections. Although our estimate here is $2800 a square foot, for the recent ATM procurement the winning price was closer to $2100 a square foot, according to an unofficial calculation I ran out in a slide rule, dividing total ATM prices by the total number of square feet that were being produced. Therefore, a 20 - 25% saving in array fabrication costs is reasonable to postulate for this model. However, the total cost that we have generated is based on $2800 a square foot for live panels. When we looked at the prototype costing, we felt that since the first time through the full production cycle, it would be a little bit more expensive, and we assumed a cost of $3500 a square foot to produce the prototype solar array, for the one live section only. For the dummies for both the prototype and for the engineering test model hardware we use the number of $1400 a square foot.

Using these numbers, in the case of the engineering test model, if you multiply the total array area that we're producing about a thousand feet by $1400 a square foot, you come out with roughly half the cost of the total cost of the engineering test model array hardware. The remaining dollars go into such things as test equipment, engineering development and factory follow and other similar cost elements. The ratio of array manufacturing costs to the total array costs goes up in the remaining elements of the program. For the prototype, the array production costs are about 75% of the cost of the array program, while for the flight hardware, the production costs are about 85% of the array costs. In summary then, this is the way that the costs break down as we developed them, and the total cost for the array again, is 16.1 million dollars.

If we again take a look at the 9.35 million dollars, which we have previously noted as the repeat cost for one flight system, figure 26 shows the way the costs break down by component. The array again hogs the lion's share of the costs, comprising about 2/3 of the total PSM costs. The remaining components are costed as the chart indicates. The Project Office is again priced in at about 5% of the total unit cost.

This completes our presentation on the cost of a complete program, and the sequence which the program would take. Repeating again, the total program cost is estimated to be 33.8 million dollars and the total program schedule will require 39 months.

Arvin Smith

We finally come to part three which is of particular interest to the office of Advanced Research and Technology. Dan Mager will talk about some of the critical technology areas that have been identified as a result of the study.

Dan Mager

What we tried to do during the course of the study program was to conceive a power supply for a manned spacecraft that has the philosophy of the power utility company behind it, designed to fit into the most general kinds of mission that one can conjure up. We wanted to keep the interfaces between the power supply and the spacecraft to an absolute minimum so that whatever the demands of the mission, the power supply could be easily integrated and also provide adequate amounts of power. The proposed configuration was shown to supply the specified loads, and further can be supplemented in a modular manner to get almost twice the amount of power that we've initially designed for, depending on the mission demands. However, even though we produced a very general power supply configuration, we appreciate that it may not be the configuration one may desire for a specific mission. Nevertheless, there are a number of development programs that can be performed at the present time to advance the technology for any design of a large power system that will go into space. In figure 27, we again see our friend who was backward last time - he's about ready to start on some of the more critical development problems which have been identified during the course of the program. Figure 28, shows that we can take a number of paths to arrive at a final configuration for the large solar array power supply in space. We can take the path which is noted as the PSM path which leads to the design of a general purpose power supply that we have described. We could examine how it fits into various missions - perhaps make some small modifications to it as missions become more defined - but the approach would be to produce a versatile, general purpose design. Another path that can be followed is shown in the right, to define a specific mission in detail and design a power supply for those specific requirements. In either event, no matter which path is to be taken, there are technical problems that are generic to any solar array power supply that you will design.

We have identified six significant problems of this general nature. Two of them are analytical, one being the dynamic analysis which has already been mentioned during the morning's discussion. The other one is enhancement of the perturbation analysis, which was examined in a simplified way during the Houston Study. The other four problem areas are of a special hardware nature that should be looked at. The first of these relates to plume effects in the array. There is an SPS engine and several RCS thrusters aboard the spacecraft. We'd like to know the effect on the array of the possible deposition of combustion products, and understand the effect of distance of the array from the various engines. I believe Houston may be performing this program at present. Another hardware problem that we consider generic to any large power system design is the development of nickel-cadmium and silver-cadmium storage cells in large capacity ratings of the order of 100-125 ampere-hours. Some work is being done to develop cells of this size by various battery manufacturers. We feel a test program is appropriate to exercise the cell according to the orbital demands that we are going to make on the cell, to run charge-discharge profiles over the anticipated life of the battery, to build up reliability history, and to ascertain the nature of the performance degradation of the cell over the lifetime of the mission. Data of this nature lends to a tighter system design and analysis of power system performance.

The last two problems have to do with the array. I've called attention to the orientation shaft as one major item of general purpose hardware to be developed. This shaft, between the spacecraft and the array, includes a major portion of the deployment mechanism, the motor drive and bearings, and the two degrees of freedom mechanisms. It includes slip rings to transfer power from the array through the two-degrees-of-freedom assembly, to the spacecraft. This entire shaft assembly can be developed prior to hard definition of a mission. The last problem, simply noted as array, has to do with two design aspects. One concerns the hinges and latches that are required during deployment of an articulated array, with perhaps the need for retraction and redeployment capability. The other important aspect is the need to develop a configuration of the solar array to operate successfully through a large number of extremely severe thermal cycles. As George stated, when we get into a very lightweight array, the temperature extremes that an array can see on a per orbit base can easily go from -150^{o}C to +80^{o}C, doing this every hour and a half in a 200 n.mi. orbit. We have found that this is a considerable design problem. I don't think that anybody has yet tested an array for a 1000 cycles to -150^{o}C. We feel that a program to build a sample array of a configuration that can survive that thermal environment is a contribution that indeed should be going on now.

Now we want to talk in some detail about each of the various problems that we had on figure 28. We noted two analytical problems, the dynamic problem and the perturbation problem. The sources of the perturbation forces are the natural forces on the solar array shown in figure 29, which will cause it to have some drag, some lift, some torquing due to the gravity gradient, some torquing due to the magnetic dipole forces and torquing due to solar pressure. If we assume that each of these torques acts on the spacecraft to disorient it from the attitude which the spacecraft wishes to have, the RCS thrusters must be fired to bring the spacecraft back to its proper attitude. We can thus relate the fuel consumption required to maintain the spacecraft positioning to the various perturbation forces. The forces are a function of the sun-vector relationship with the spacecraft axes, the alignment of the spacecraft in its orbit and the instantaneous alignment of the magnetic flux. We feel that what should be done is to build a mechanized program to examine in detail and catalog the various perturbation effects and fuel penalties as a function of all the mission variables. One of the important variables will be the accuracy desired from the spacecraft attitude control system. We would like to select a spacecraft attitude accuracy, and determine how often we have to fire the RCS thruster to compensate for the torques that the natural forces introduce. Looking at figures 30 and 31 which summarize the important variables of the problem, certainly the altitude of the spacecraft and the inclination of the orbit will have an impact on the drag, the lift, the magnetic dipole, and the gravity gradient. The sun elevation is a seasonal variant which sets the relationship between the solar paddles and the spacecraft on a particular day. The array area and aspect ratio are going to affect all of the perturbation forces. For example, what is the optimum aspect ratio to select for the various altitudes, for the various inclinations, and for various attitude accuracy requirements? Certainly a second of arc pointing accuracy for the spacecraft causes a signficant design impact compared to a 1 degree accuracy requirement. These factors then should all be put into one computer program, so that given the requirements of a specific mission we can calculate the fuel consumption. For the general case we feel that it is important to mechanize the problem to produce answers for whatever mission or system study requirements are generated. This is then the first of the problems. We would like to point out again that it should and can be done now, in the most general case to suit a wide variety of mission needs.

Slide 32 shows a proposed perturbation analysis program, approximately 5 months long and which we have laid out as follows. The first period of time would be used to develop the force equations, using general parameters rather than specific numbers. The next step would be to mechanize the equations. Finally we would run the program on sample problems to demonstrate its application, and then produce the final report. Figure 33 summarizes the fruits of the program, in our cornucopia. Basically the output from the program can be summed up as, tell me what mission you are going to fly and I'll tell you your fuel consumption and your fuel comsumption rate per orbit, per day, per month. We can select an attitude correction duty cycle we choose to have and find exactly what attitude disturbances we have which we are going to have to correct by firing the RCS thruster. For example, is it better for the spacecraft to drift from its desired attitude by 2 degrees and fire the engine or is it better to let it drift off 1/2 degree and then fire the engines? Which duty cycle is going to give the minimum fuel consumption? What is the array size and the shape that will give the minimum fuel consumption, the highest spacecraft pointing accuracy, the slowest movement of attitude change?

Let's look again at figure 28. We have spoken about the perturbation analysis - we would like to spend some time discussing the dynamics of the system. We have perturbation forces that can cause attitude pointing errors in the spacecraft which require correction. We have docking forces that can produce pointing errors. We have movements of the astronauts perhaps, and firing of the SPS engine to change the orbit or to correct for drag. All these things represent significant transients to the spacecraft which must be corrected for by the firing of the RCS engines to get us pointing back in the desired direction. Figure 34 outlines the three modes of dynamic analysis that should be looked at. The first mode is the initial deployment of the solar array. We've launched the spacecraft and we have to get the paddles deployed. Are the astronauts aboard or will they come aboard later? What shall we do with the spacecraft? Will we permit the spacecraft to tumble while we are deploying or do we want to hold it steady, perhaps by momentum wheels, while we deploy? What does it cost to go to each of these alternatives in fuel comsumption and in forces that the array will see? How much mechanical strength do we have to add to the system to accommodate a possible tumbling while deploying - which is not really something that we wish to do. And the basic problem is, how stable is the system while we deploy? Can we indeed build a practical servo mechanism considering the dynamics of the spacecraft, so that the system can be stabilized when we have finished deployment?

The second mode of operation is steady-state flight. The deployment is completed and we are stable. Now occasionally, at some duty cycle, we fire the RCS thrusters, or we dock, or the astronauts move about in the spacecraft, causing disturbances. In any event we generate a pointing error which requires correction. As we repoint the spacecraft, the paddles will move to keep pointing at the sun. Are we stable under those conditions of operation? What is it that we have to do to the servo designs to stabilize the system for that mode of operation.

The third mode of operation that should be analyzed is retraction and redeployment of the system, which we have represented in the lower part of the figure. You will note that the retracted configuration need not be the same configuration that we had at the point of initial deployment. The retracted configuration may therefore represent a different reflected inertia and compliance to the spacecraft control system. So again we have several complex servo parameters to look at to ascertain if the system is stable. Redeployment from this state may therefore also be different from the initial deployment, in terms of compliances, inertias, and stability. So again we would generate design criteria for the array to meet. Figure 35 summarizes the dynamics of the spacecraft and the large array. The spacecraft has three degrees of freedom of motion, defined by the red lines with the black tips. The spacecraft can rotate about any of these three axes or around all three at the same time. We have defined the panel system as having two degrees of freedom, about the yellow axes. One axis is common to both paddles, and each paddle has a second axis of its own which is normal to the common axis. For the purpose of graphical illustration, it was decided to mount the sun sensor to the spacecraft with two degrees of freedom with respect to the spacecraft. The sun sensor telescope is shown out of scale for purpose of graphical clarity. Of course, we don't have to mount the sensor to the spacecraft. The sun sensor can be mounted at the base of the boom, at the outer tip of the boom, at the base of the paddle at the outer tip of the paddle - it can be mounted at whichever point is convenient for the mission. Each of these mounting places has some advantages and disadvantages which we will not elaborate on at the present time. For our example, the sun sensor shown is mounted to the spacecraft and has two degrees of freedom about the green axes.

Let's now fire the RCS thrusters to correct the attitude of the spacecraft. We will then disturb the sun sensor from its sun pointing position. The sensor will move back to the sun vector about the green axes. When the sensor has relocated the sun again with respect to the spacecraft axes, the array will then orient itself toward the sun. Figure 36 is a simplified block diagram of the motions that I have just described. We have taken one axis each of the spacecraft, the sun sensor, and the array, and shown it on the diagram of figure 36. In the final analysis, this diagram must be interlocked with three axes on the spacecraft, two axes on the array and two axes on the sun sensor, remembering that we can also mount the sun sensor directly to the paddle and have the panel drive supply the degrees of freedom for the sun sensor as an optional choice.

Let's now assume one axis motion of the spacecraft. The rectangles with the diagonals represent resolution points in our feedback loop. The circles with the diagonals represent summing or subtraction points. Motion of the spacecraft about one axis will displace both axes of the sun sensor from the sun vector, in the general case, shown as two outputs from the resolution point. The resolved components of the other two axes of motion of the spacecraft are shown entering the summing point. The output arrow of the summing point represent the total resolved motion of a single axis of the sun sensor. We must now move the sun sensor back to regain the sun. The angular relation between the sun sensor and its axes of rotation enters resolution point 2, which is an analog computer that relates sun vector, and spacecraft axes, and provides array drive signals. The array drive signals are shown as an input vector to the block representing the array drive servo.

While being driven to its new position, the array is going to generate a reaction torque, which is resolved at resolution point three into three components about the three axes of the spacecraft. The reflected reaction torque closes the dynamic interaction feedback loop. The total diagram thus is a simplified illustration of the dynamic problem. Remember again, this diagram should be expanded into three axes for the spacecraft, two for the sun tracker and two for the array orientation. Having generated a general problem solution, and given parameters which describe the mission, we can go ahead and describe the parameters which define the dynamic responses of the solar array. We can calculate the effect of variations in the array design parameters on the spacecraft fuel consumption. And given mission parameters we will have a tool with which to understand the impact of variation in the parameters. Figure 37 shows a proposed schedule for the dynamics program. We feel that this program requires about 8 months, provided we do the perturbation program that we previously discussed. If you don't do it first then the perturbation program is a necessary part of the dynamics program and the total schedule must be extended by about 2 months. The mission and spacecraft constants would be evaluated parametrically, although the program can be done more simply for a specific problem. In the latter case, however, it becomes less useful for the more general case. After establishing the mission and spacecraft parameters, we would define the array parameters which we wish to investigate, we would then develop tables of equations, perturbations, and array inertia parameters. As the spacecraft flies in its orbit, and the paddles seek the sun, the inertias reflected to the various axes of the spacecraft are going to change periodically. We must develop tables of the variables to pump into the program. After deriving the equations of motion, we would evaluate the deployment mode of operation, the steady-state flight mode of operation, and the retraction and deployment mode, to see if we are stable, or what has to be done to establish stability, either mechanically, or by use of electronic damping networks. A final report would be issued which covers the study results.

What do we expect to get from such a program? Figure 38 again summarizes the fruits of the program. An important output would be the specifications for the array and boom designs. How stiff can we make the array? How lightweight should it be for the various kinds of missions? With a need for one second pointing accuracies, there will be a different array design than one designed to be compatible with one minute or one degree pointing accuracies.

We will define the sun tracker requirements. Where shall we put the sun tracker? If we place the sun tracker at the end of the array right at the very tip, every time you are going to try to seek the sun this flexible paddle will flap. Is that a desirable place to put the sun tracker? If you place it on the spacecraft, you have to provide 2 degrees of freedom. In addition, there is a high probability that you will shadow the sun tracker some of the time and you may need two of them, with logic network to switch between alternate sun trackers. Intuitively it would look like the tip of the boom is the best place to put it, but we ought to prove that before we make a decision.

We would like to take a look at the array reactions - when we have transient vibrations, when we are trying to stabilize the overall servo loop, when subjected to forces when we slew the paddles to a new position, and when correcting for the dynamic disturbances. Some of these forces may indeed be more important than launch forces to the array design, especially if we get resonance buildups.

We will also get a motion picture of the way the system moves every time we fire an engine or deploy, including how long the paddles will flap, how they affect the spacecraft, and other important factors. We think this is a significant result. We have spoken a number of times about the spacecraft attitude stability in the steady-state mode. Every time we fire the RCS engine to correct the attitude, is the system stable and what do we have to do to make it stable? Shall we let the spacecraft drift off 1° before we fire the engine, $1/2^{\circ}$ or what is a reasonable choice to minimize fuel consumption. This then is the dynamic program. It can be solved as a general problem, and not in the specific PSM or other system configuration. Solution to the dynamic problem is included in the Phase I program that Mr. Barna showed you previously.

Figure 39 summarizes the four hardware problems that we recommended be done along with the two analysis problems. Let's take a look at the large capacity battery test program. We have stated that we need 100-125 ampere-hour cells to install batteries aboard the spacecraft efficiently. With a smaller cell, there are large weight penalties due to packaging, mounting and so on. We recommend that storage cells be assembled in an appropriate configuration on the test bench, and a simulated mission lifetime battery program be run. A six month to one year battery cycling program should be performed to find out the manner of degradation of the batteries and their charge and discharge characteristics, to generate criteria for the design of the electronics of the power supply system. A consistent problem that we always get into with a new cell is the proper design of the electronics to control battery charge and discharge. Properly designed electronics should be designed for the end of life characteristic of the battery, which can only be derived from a good long program which simulates the life of the mission as an ideal, alghough it can be shortened some. You can't speed up the life test of the battery by rapid-cycling, because you change the parameters of the battery. A proper characterization program is something then that can and should be started now.

The second hardware area, the thruster plume effects, we spoke about in passing before, to evaluate the effects of the engine effluents on solar cells in proximity. We believe Houston has already begun this work.

The third area, the low temperature array configuration, we also spoke about previously, principally concerned with developing a good low temperature solar array configuration.

The solar array development model is concerned with developing hinges and latches, and the orientation shaft and its mechanisms. Let's just dwell on these for a short period of time. Figure 40 shows the major elements of the array systems, including the deployment mechanism, a boom, an orientation drive with 2 degrees of freedom, and a power transfer mechanism that takes the power from the array through the double rotating joint to the spacecraft. Looking at the array panels, these include the substrate design, the hinges and latches necessary to interconnect the panels in a plane as well as deploy the panels from the folded assembly, and which can be designed to permit you to retract if the need arises, and finally bus wiring and solar cell connections. Now the total program, shown in figures 41 and 42 can be undertaken completely or in part, such as separate programs to develop the low temperature configuration or to develop retraction techniques and mechanisms. Or we could build and test a simple array section which would contain hinges, latches and the various array components. We could also make minor test articles of a hinge, a latch, the slip ring wiper system, bearings, the two degrees-of-freedom drive mechanism and build and test them in various programs. Ultimately, however, at least an abbreviated model of the total solar array should be constructed as an engineering test model including the above component parts and also the complete power transfer mechanisms and be submitted to an environmental test program. Having the results of the total program, we could then prepare the total specification of the hardware designs needed to fly a large solar array in space for one year. We feel that the extent of the total program is 18 months. Again the whole array development as described here need not be done as one complete package, but we recommend that a total program be considered even if the work is spread over some period of time. With that we conclude our portion of this morning's discussion and we will be glad to answer any questions that you might have.

Question and Answer Period
following Presentation by George Barna

NOTE: The following statements are not quoted, but paraphrased for clarity.

1. Q. Is the 100 #/KW power out of the array or daytime power?

   A. Array power - 100 #/KW or 15 KW capability weighs 1500#. 9.7 KW daytime load and 5.6 KW for nighttime load.

2. Q. Did we go through a structural array design?

   A. We stressed out array structure for launch and deployment. Array weighs 0.81 $\#/ft^2$. We assumed a rigid array. Dynamic requirements were beyond program scope.

3. Q. How much of the system weight is necessary to match system to voltage variations?

   A. No weight specifically associated with voltage matching. System is capable of operating over a wide voltage range. Converters are used to extract specific required voltages from system input variations.

   Q. Why not orient to regulate power capability?

   A. Reliability of this type of tracking system scares us.

4. Q. If you have a manned system, will you not have large power variations? Is it easy to compensate?

   A. Yes - variations will be large, compensation is not difficult. Primarily a battery design problem.

5. Q. Have you looked at an upside-down cycle for better life, or deeper depths of discharge?

   A. No - we stuck with things we were sure of.

6. Q. How does weight compare to an unmanned spacecraft?

   A. Manned increases weight by 10-15%.

7. Q. Can we use astronaut?

   A. Yes - but not until the astronaut's capability in space is better understood.

8. Q. Does weight bookkeeping include the array pointing system?

   A. Yes - under the category labeled D&O (Deployment and Orientation).

9. Q. Define the sun angle.

   A. Angle between orbital plane and sun vector.

10. Q. Does the 100 #/KW include compensation for docking forces? Was this the limiting force?

    A. Yes, it does include compensation for docking, but we found SPS engine firing to be the maximum force. There is no penalty for docking. We considered retraction, but found that favorable array orientation would be adequate to withstand the SPS forces.

11. Q. Is the SPS used for drag make-up?

    A. Yes - for drag, altitude, orbit plane.

12. Q. What is the difference in specific impulse between SPS engine and RCS thrusters?

    A. Not much - 300# for the SPS and 200# for the RCS.

Question and Answer Period
following Presentation by Dan Mager

NOTE: The following statements are not quoted, but paraphrased for clarity.

1. Q. What do you mean by shock for this case?

   A. The various acceleration forces occurring in flight such as docking, SPS engine firing, and other transient motions, possibly astronaut movement.

2. Q. Why do you need a sun sensor? You know where the sun is on a daily basis, and you know the spacecraft orientation from its attitude control devices. The sun position can be computed.

   A. That's true. It doesn't matter how sun orientation is related to spacecraft axes. Computation is as adequate as sensing. O.K. I'll accept computation.

3. Q. Is there a way to sense the power that the array is giving out at any given time?

   A. The Maximum Power Point Tracker senses the power output, and finds the maximum power point in voltage-current characteristic. When the battery voltage, as a function of temperature indicating full charge is sensed, the array output power is reduced by changing the operating point on the I-V curve to a lesser output power. In other words, when not needed, the power from the array is not used.

   It is not mechanical tracking, offsetting array from sun vector normal, but electronic tracking.

4. Q. Did we consider artificial gravitation?

   A. No.

5. Q. Have we flown maximum power point seeker?

A. We have a qualified model which is not part of any specific system. MPPT doesn't solve all problems. Primarily useful when large variations in array parameters occur due to array thermal profile or radiation degradation. It reduces spacecraft heat dissipation problems and provides extra load capability at beginning of life.

6. Q. How do we sense maximum power?

A. Measure current and voltage and analog compute along power curve.

Q. Manual tracking available?

A. No.

Q. How often is the array power varied?

A. Continuously with high frequency impedance matching switch.

Q. Is a capacitor needed?

A. Yes - for filtering effect.

7. Q. What can you report on solar array cell interconnections?

A. Lunar Orbiter $\pm120^{o}C$, moly strip cycled 600 times.
Nimbus to $-90^{o}C$, 1400 cycles, copper strip.
Other programs to $-100^{o}C$, 6-700 cycles with silver mesh.
Below $120^{o}C$ array temperature, significant design work is needed.

8. Q. On the shadowing program you performed by counting shaded squares, isn't the shadowing loss more than an area effect?

A. Yes and no - For a single cell shadowed in a single string, the entire string is lost. When cells are connected in series and parallel combinations, shadowing loss approaches percentage of area shadowed. Our tests indicate that one cell shadowed when 6-8 cells are connected in parallel approaches shadowed area ratio.

9. Q. Did you have any concern for electrostatic build-up on array?

A. No - not part of program. Certainly would be a matter of concern in a hardware design program.

10. Q. How do you bond to substrate for the $\pm 120^{o}C$ array.

A. RTV adhesive - proprietary information.

11. Q. Did you do tradeoffs on 3^{D} degree of freedom to eliminate shadows?

A. Not specifically, but spacecraft need be rolled only 6^{o} in 3^{D} degree of freedom direction to eliminate shadow loss.

Q. Can power module be rolled separately?

A. Yes.

12. Q. Why did you configure the solar paddles in the shape you've shown? Why not similar to the aspect ratio of airplane wings?

A. We performed many tradeoffs on various shapes. The shape you indicate would result in interference between array and spacecraft during certain portions of the year.

13. Q. Do you have enough experience to recommend slip rings?

A. The Nimbus satellite has successfully used slip rings for more than a year.

14. Q. Have we considered other than slip rings, such as rotary transformers for the power transfer joint?

A. Yes - but we considered these other mechanisms not state of the art.

15. Q. What would the aspect ratio be if array became twice the area indicated?

A. We didn't investigate all ramifications precisely, but we'd try to keep the same aspect ratio. The specific ratio would result from a trade-off study.

16. Q. Do arrays of this size make solar cells cost more, difficult to get?

A. There may be some temporary price transients in the market, supply and demand would work, but our experience has been that the higher the demand the lower the price. There are some possible new manufacturers who might go into the solar cell business if the situation was right.

17 Q. What attitude of array is used during orbital night?

A. A slewing memory is used to maintain approximate correct array orientation position.

SOLAR CELL TECHNOLOGY - WELL DEVELOPED AND UNDERSTOOD

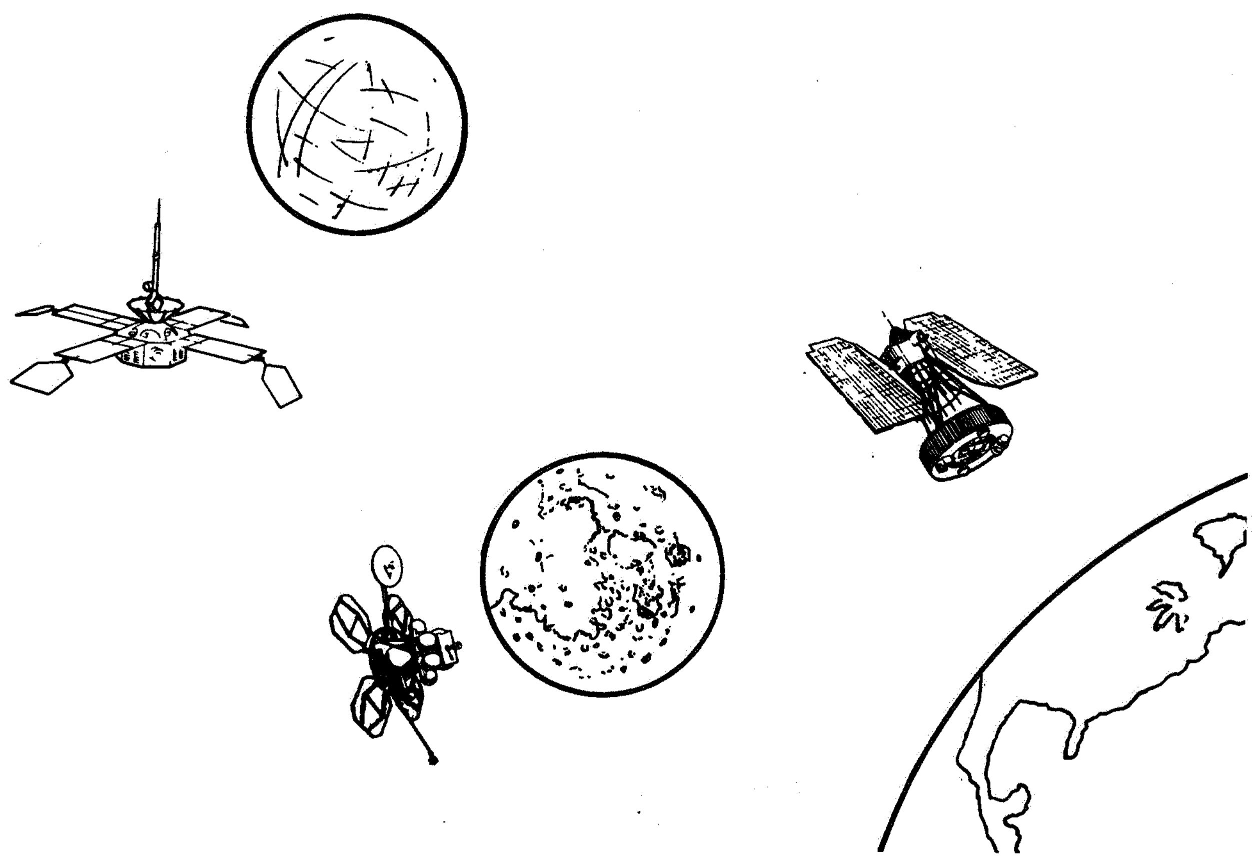


Figure 1

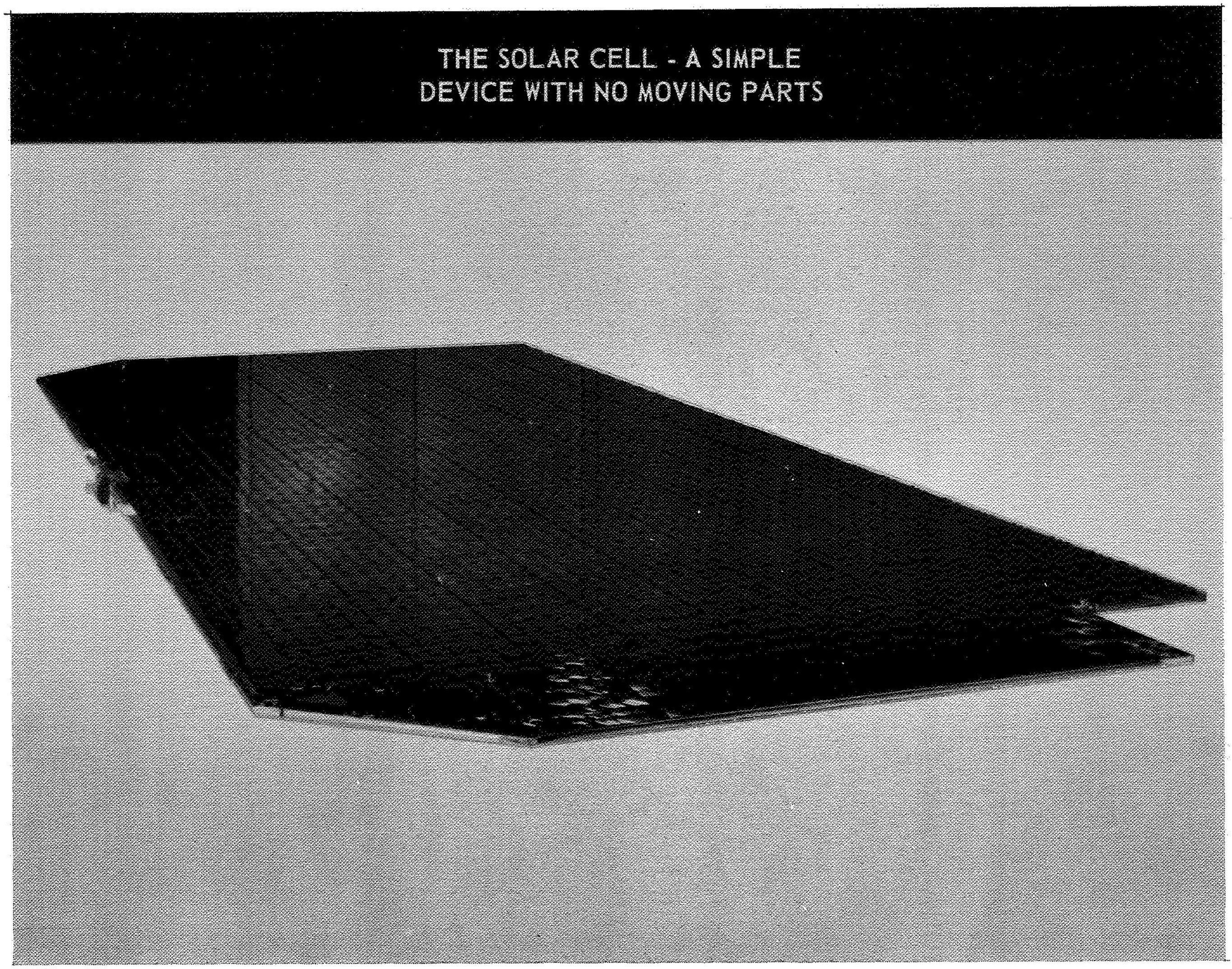


Figure 2

SIZE CAN BE A PROBLEM

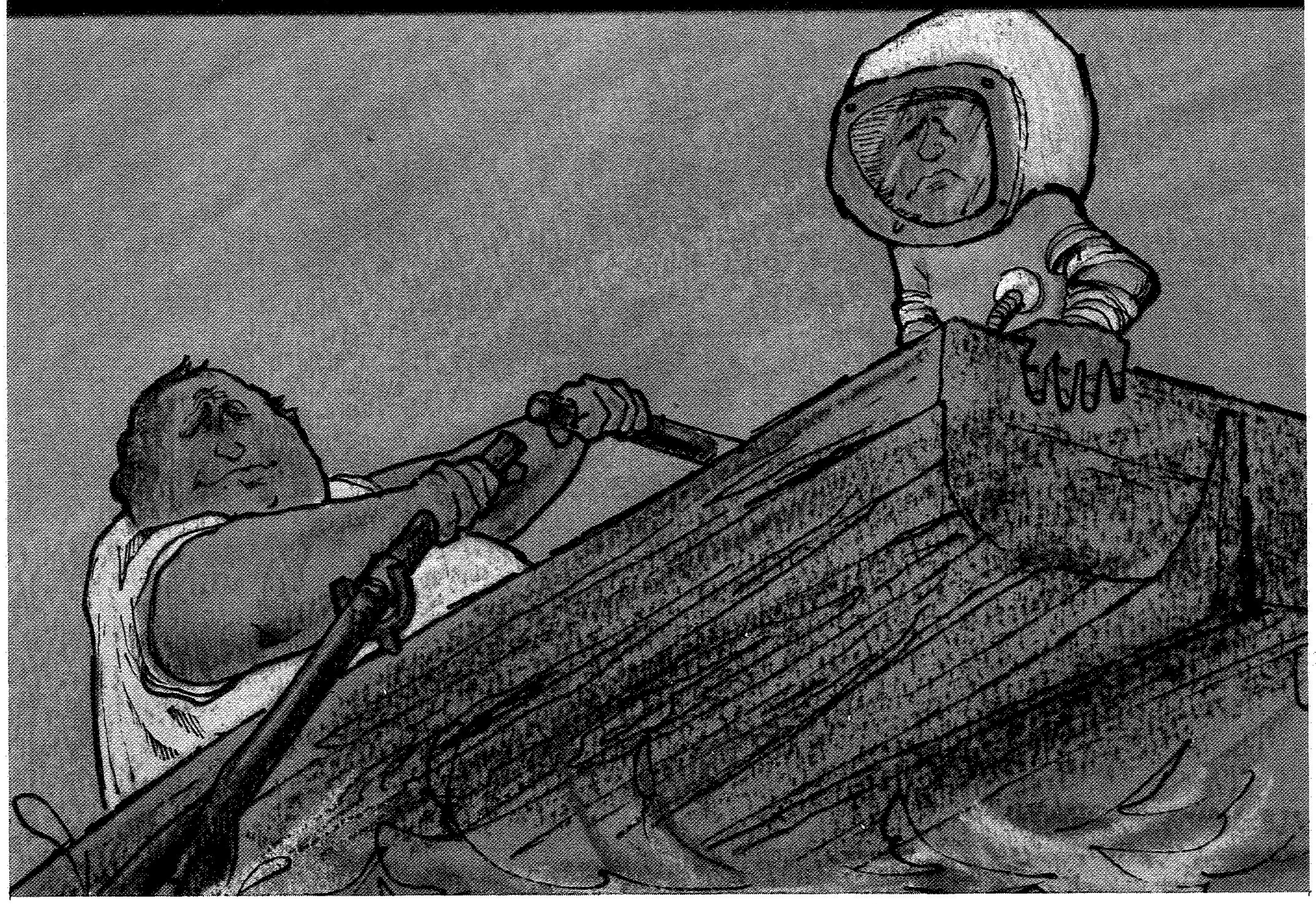


Figure 3

SIZE CAN BE A PROBLEM

- We are concerned with size reflected as:

  AREA -- WEIGHT -- COST

- Therefore, we must:

  -- ORIENT, to minimize size

  -- DEPLOY, to trade launch volume for in-orbit area

  -- PAY FUEL PANALTIES, to compensate for perturbations

  -- DESIGN TO MINIMIZE INTERACTIONS, static and dynamic

- But all of these problems can be handled!

Figure 4

DEPLOYMENT OF LARGE AREAS IN SPACE

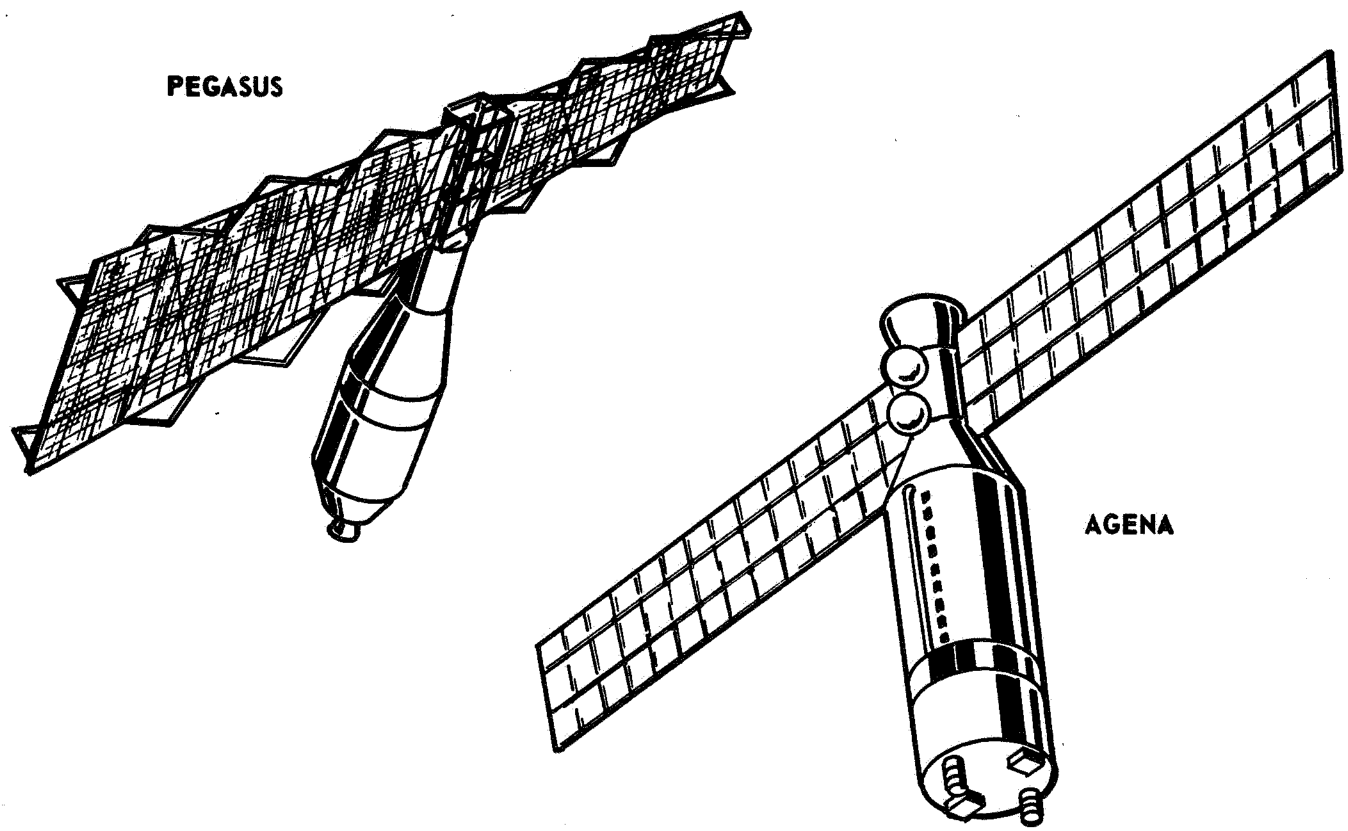


Figure 5

IDENTIFICATION OF TECHNICAL PROBLEMS

DESIGN CONSTRAINTS

- ADAPTABILITY OF DESIGN APPROACH TO ORBITS
  - 200 NMI AND SYNC ALTITUDES
  - 30 DEG TO POLAR INCLINATION
- ONE-YEAR RELIABILITY GOAL OF 0.995
- S/C CONSTRAINED TO OPERATE "BELLY-DOWN"
  - ROLL AXIS PARALLEL TO VELOCITY VECTOR
  - YAW AXIS ALONG LOCAL VERTICAL
- IN-ORBIT THRUSTING FOR ORBIT PLANE
  - ATTITUDE CORRECTION
- FLIGHT VEHICLE ENVELOPE CONSTRAINTS

Figure 6

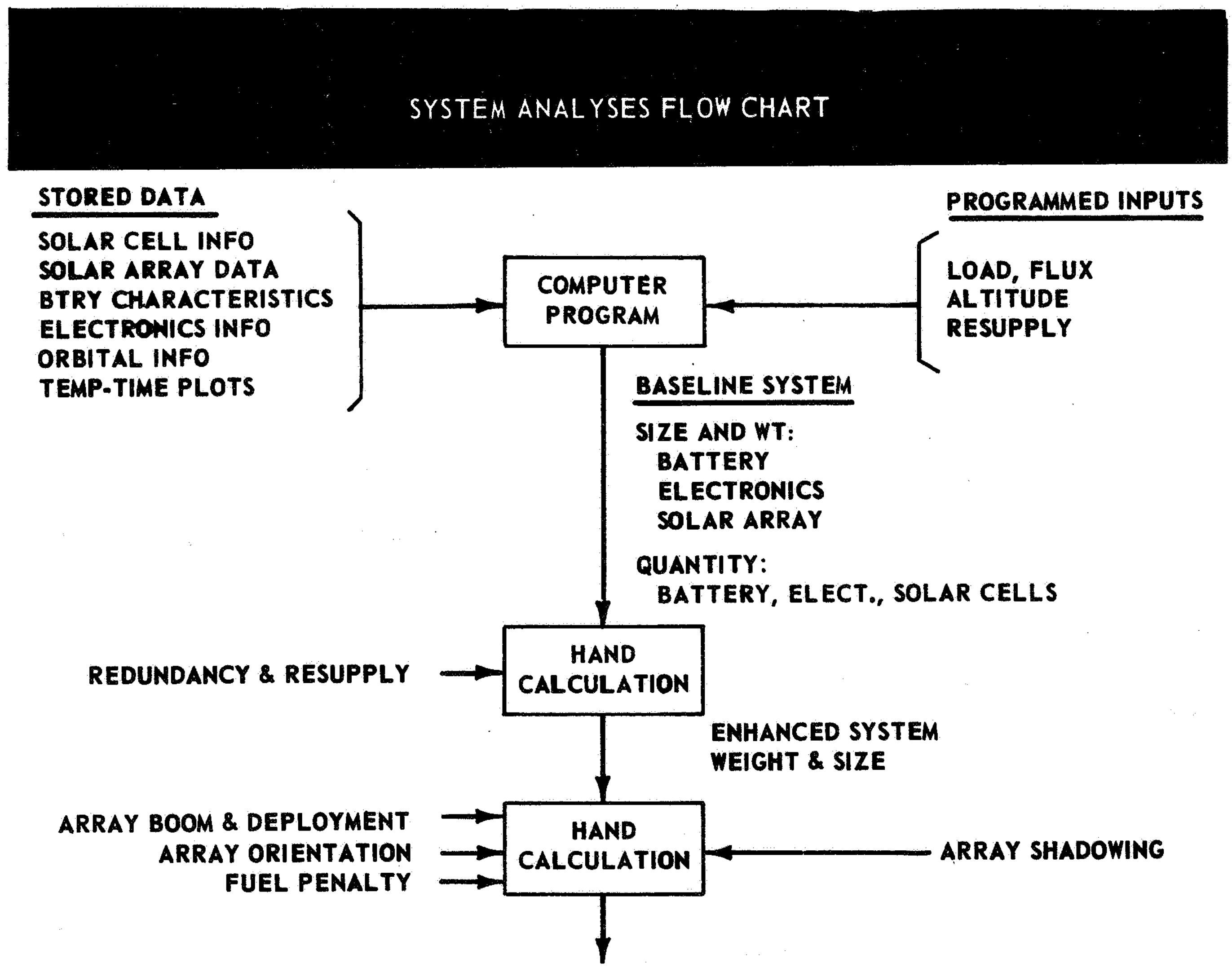
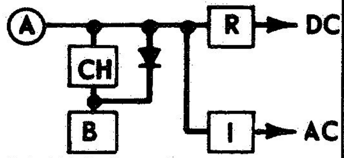
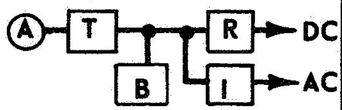
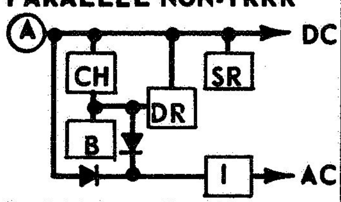
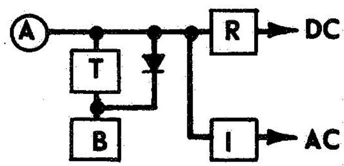

SYSTEM ANALYSES FLOW CHART
STORED DATA
SOLAR CELL INFO
SOLAR ARRAY DATA
BTRY CHARACTERISTICS
ELECTRONICS INFO
ORBITAL INFO
TEMP-TIME PLOTS
COMPUTER PROGRAM
PROGRAMMED INPUTS
LOAD, FLUX
ALTITUDE
RESUPPLY
BASELINE SYSTEM
SIZE AND WT:
BATTERY
ELECTRONICS
SOLAR ARRAY
QUANTITY:
BATTERY, ELECT., SOLAR CELLS
REDUNDANCY & RESUPPLY
HAND CALCULATION
ENHANCED SYSTEM
WEIGHT & SIZE
ARRAY BOOM & DEPLOYMENT
ARRAY ORIENTATION
FUEL PENALTY
HAND CALCULATION
ARRAY SHADOWING

Figure 7

SYSTEM CONFIGURATION ANALYSIS (5-KW LOAD)

SYSTEM CONFIG	REL EFF. 200 n mi	REL EFF. 19380 n mi	BASE-LINE REL	MODULES FOR 0.995 REL	DISSI-PATION	BUS V EXCUR.	REDSGN FOR NEW MISS.
SERIES NON-TRACKER 	0.95	0.93	0.43	17	HIGH	LOW	YES
SERIES TRKR 	0.94	0.85	0.53	16	LOW	HIGH	NO
PARALLEL NON-TRKR 	0.99	1.00	0.38	46	VERY HIGH	VERY LOW	YES
PARALLEL TRKR 	1.00	0.93	0.42	17	LOW	HIGH	NO

A = ARRAY B = BATTERY CH = CHARGER I = INVERTER R = SERIES REGULATOR
SR = SHUNT REGULATOR DR = DISCHARGE REGULATOR T = TRACKER

Figure 8

BASELINE SYSTEM ANALYSES

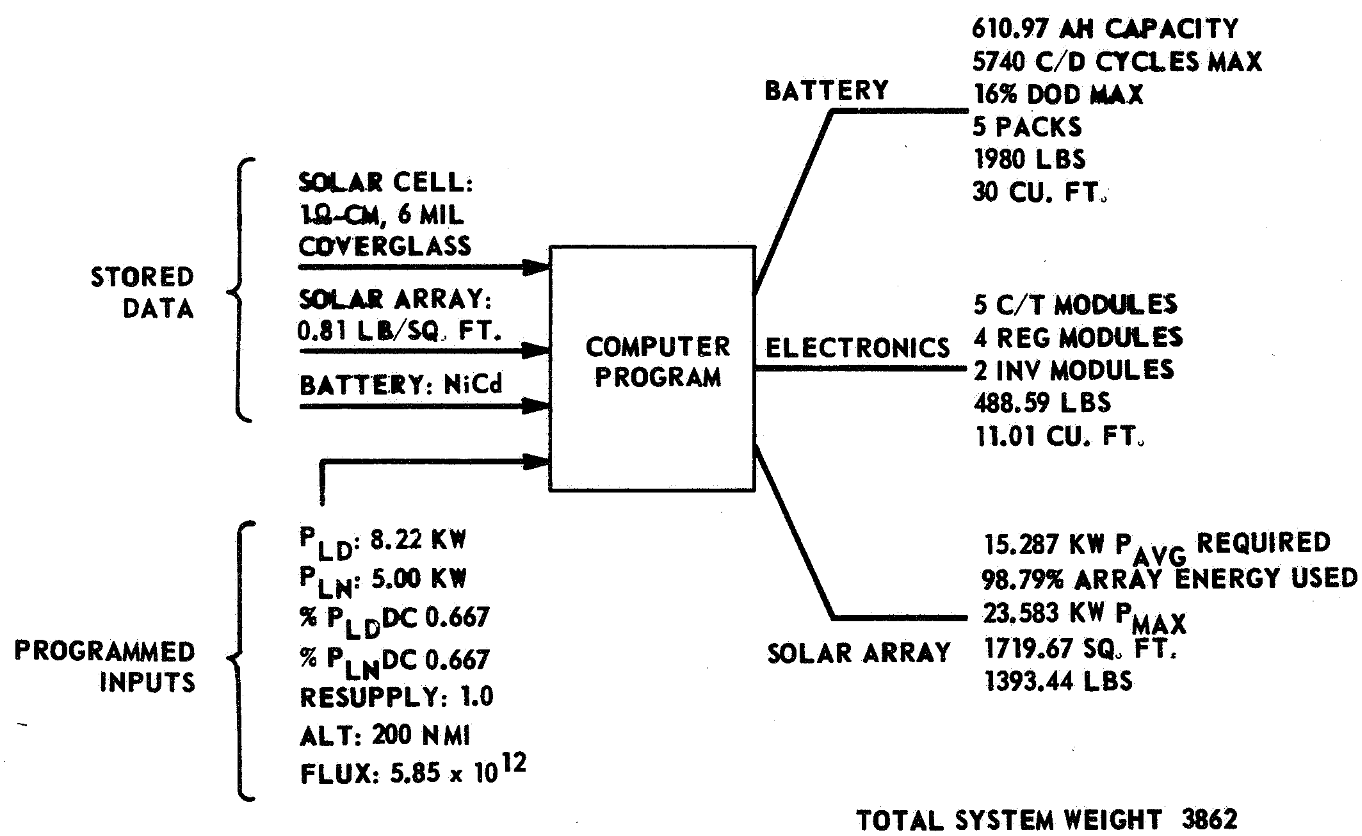


Figure 9

RELIABILITY ENHANCEMENT OF BASELINE SYSTEM ANALYSIS

	REL (1 YR)	WT AT LIFT-OFF (LBS)	WT OF SPARES (LBS)	RESUPP* WT (LBS)	ARRAY AREA (SQ FT)	BATT TYPE	EQUIP. COMPLEMENT MODULES**			
							BATT	REG	INV	C/T
BASELINE SYSTEM	0.424	4026			1720	NiCd	30	4	2	5
ENHANCED SYSTEM NO RESUPP	0.995	4740	714		1720	NiCd	33	6	7	8
ENHANCED SYSTEM 45-DAY RESUPP	0.995	4280	714	360	1720	NiCd	33	6 5	7 3	8 6
ENHANCED SYSTEM MIN LIFT-OFF WT 45-DAY RESUPP	0.995	3171	251	3065	1720	AgCd	18	5 5	4 3	6 5

* 1-YR MIN

** LOWER NUMBER IS "LIFT-OFF" COMPLEMENT

Figure 10

SPACECRAFT AND DEPLOYED ARRAY

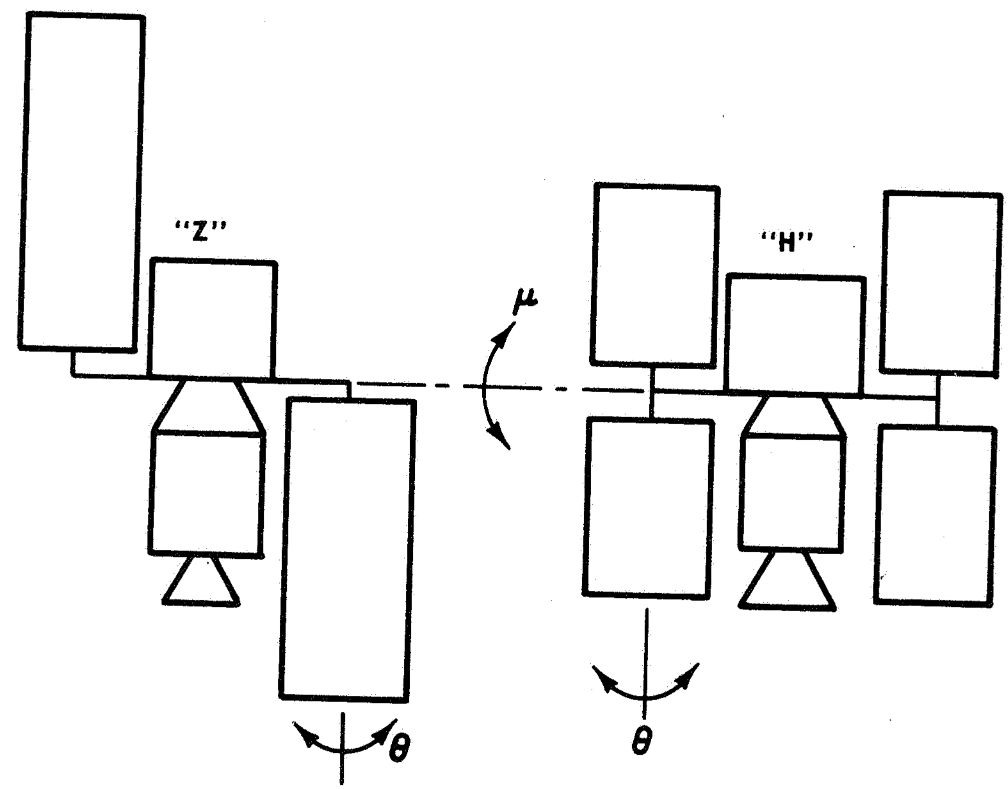


Figure 11

SOLAR ARRAY CONFIGURATION STUDIES

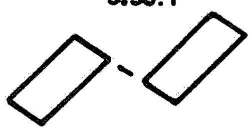
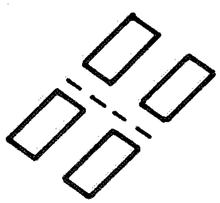
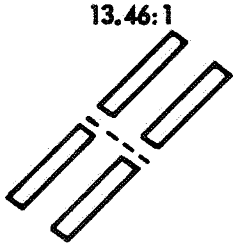
ARRAY LENGTH TO-WIDTH RATIO	AERODYNAMIC		GRAVITY GRADIENT		SOLAR PRESSURE		MAGNETIC DIPOLE
	DRAG	LIFT	PITCH	ROLL	PITCH	OTHER	
3.36:1 	1823	460* 0	1400* 0	910	9.9 8.6	6.5 3.1	38.4* 16.5
3.36:1 	1823	460* 0	313* 0	0	9.9 8.6	6.5 3.1	38.4* 16.5
13.46:1 	1823	460* 0	1400* 0	0	9.9 8.6	6.5 3.1	38.4* 16.5

Figure 12

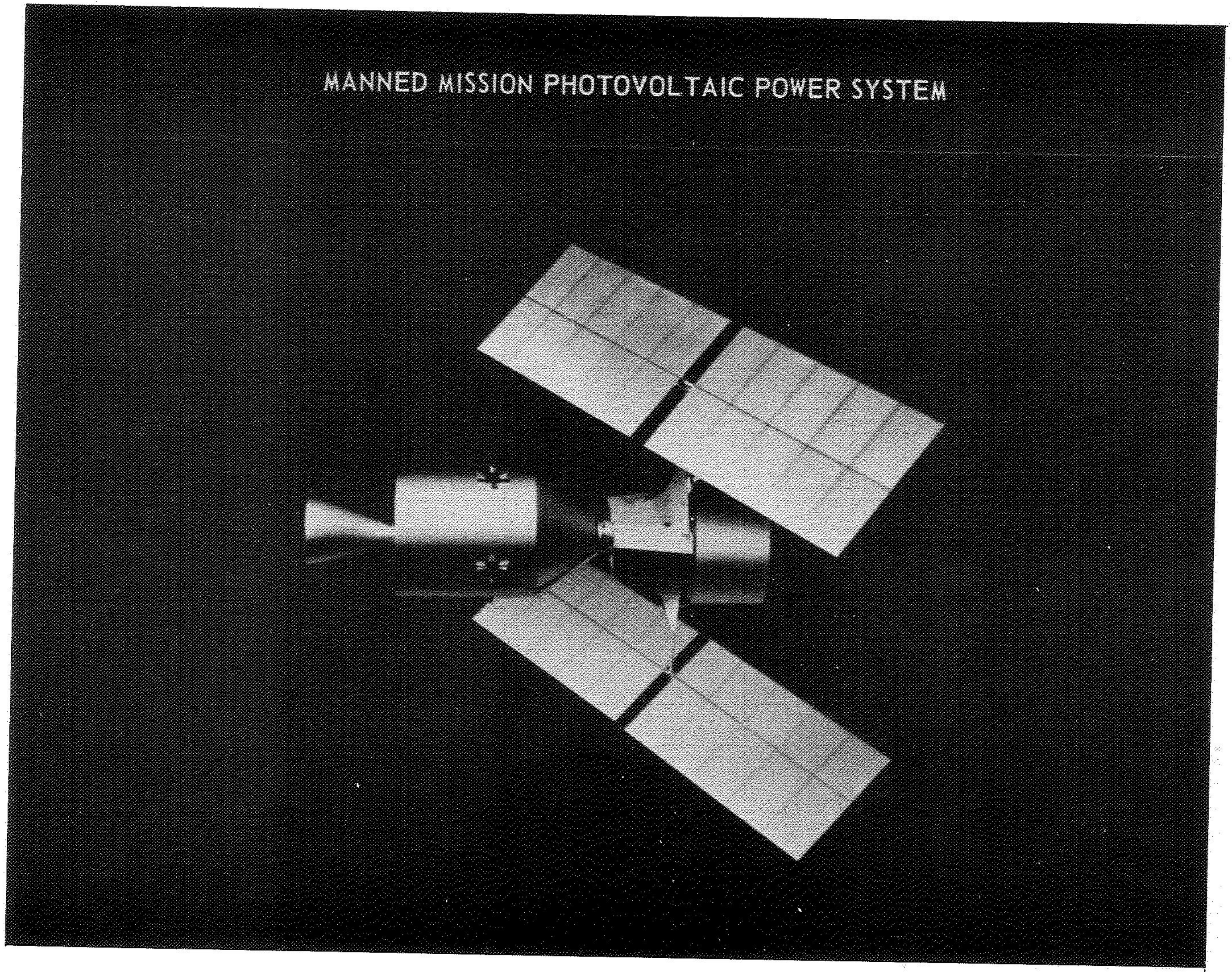


Figure 13

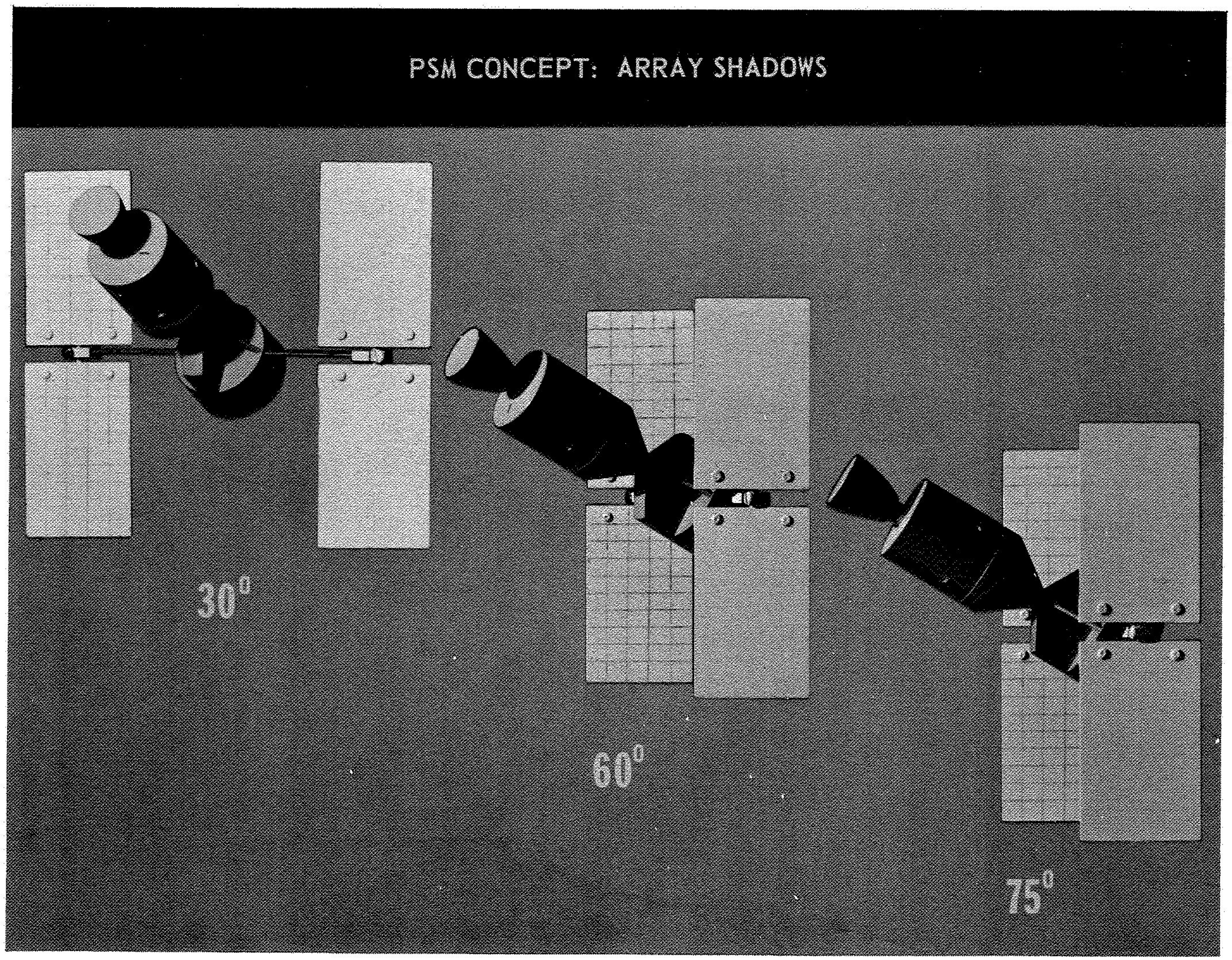


Figure 14

ARRAY SHADOWS CONFIGURATION STUDIES

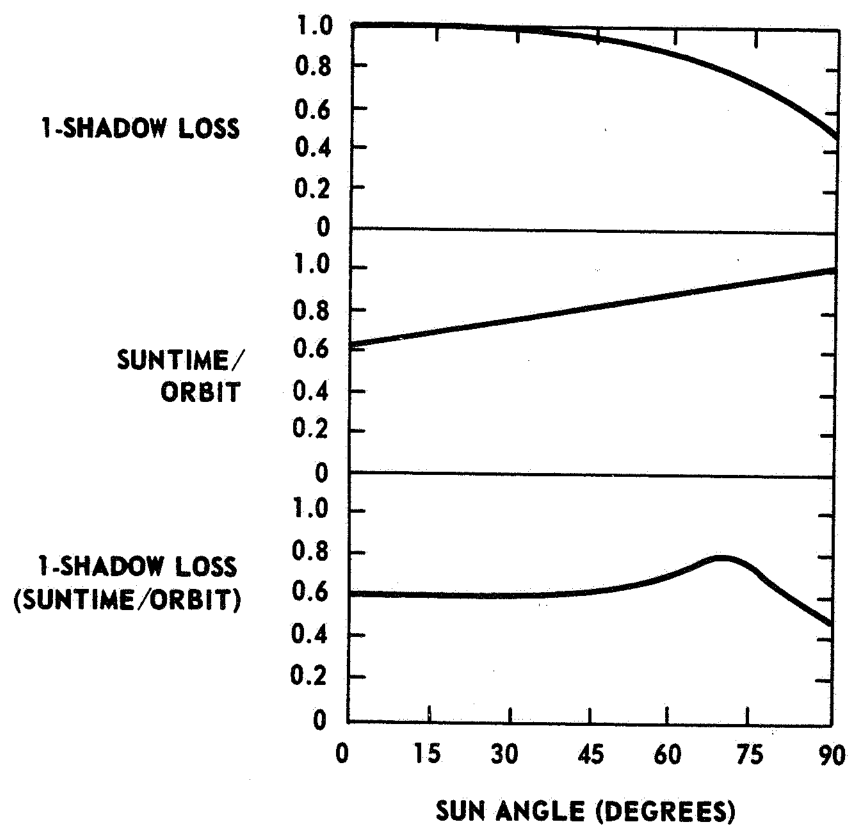
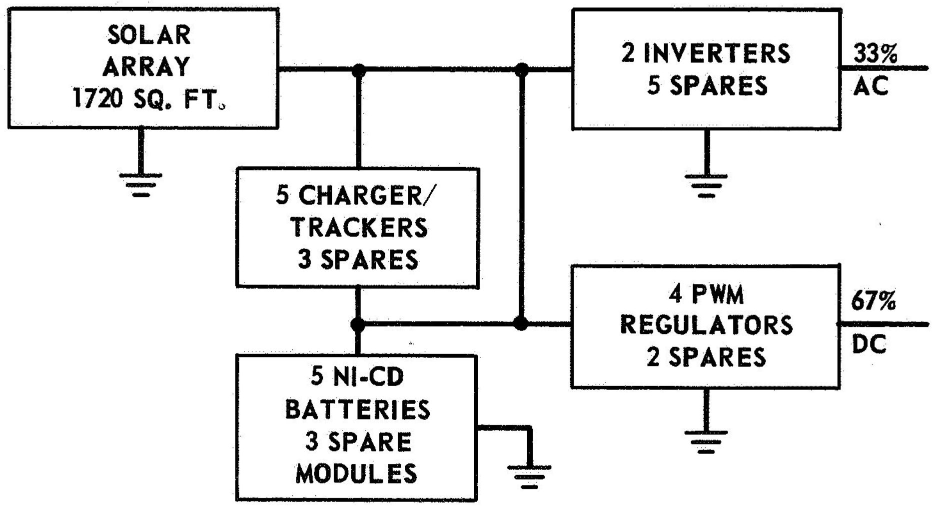
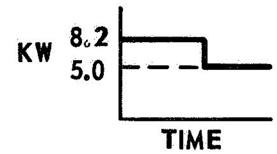
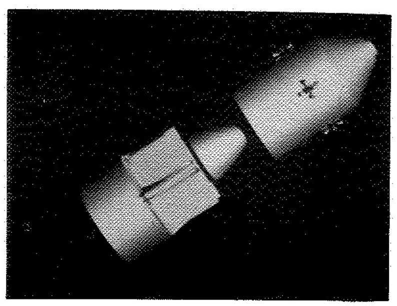


Figure 15

SELECTED SYSTEM CONFIGURATION



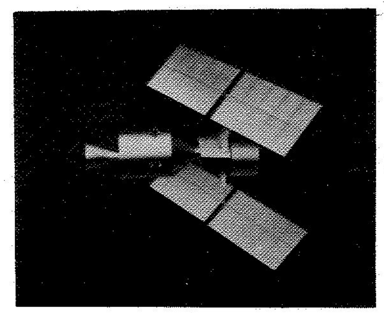




LAUNCH CONFIGURATION

PSM WEIGHT

SOLAR ARRAY	1393
BOOM & DEPLOY	104
ORIENTATION	60
BATTERIES	2178
ELECTRONICS	1005
FUEL PENALTY	1868
TOTAL (LESS STRUCTURE)	6608 LBS



ORBITAL CONFIGURATION

Figure 16

SYSTEM WEIGHT ANALYSIS

DESIGN WEIGHT DENSITIES (LBS/KW)		DAYTIME	NIGHTTIME	CONDITIONING
ARRAY	100	X	X	
BATTERY	390		X	
CHARGER/TRACKER	45		X	
POWER CONDITIONING	90			X
FUEL PENALTY	125	X	X	

DAYTIME LOADS
9.7 KW X 225 LBS/KW
=2200 LBS

NIGHTTIME LOADS
5.6 KW X 660 LBS/KW
= 3700 LBS

POWER CONDITIONING
8.2 KW X 90 LBS/KW
= 740 LBS

FIGURES OF MERIT
(W/LB)

DAYTIME: 2.8

NIGHTTIME: 1.1

SYSTEM: 1.1 (W/FUEL)
1.5 (W/O FUEL)

Figure 17

PROJECTED SYSTEM WEIGHT ANALYSIS

DESIGN WEIGHT DENSITIES (LBS/KW)		DAYTIME	NIGHTTIME	CONDITIONING
ARRAY	50	X	X	
BATTERY	200		X	
CHARGER/TRACKER	45		X	
POWER CONDITIONING	90			X
FUEL PENALTY	125	X	X	

DAYTIME LOADS
9.7 KW X 175 LBS/KW
= 1700 LBS

NIGHTTIME LOADS
5.6 KW X 420 LBS/KW
= 2300 LBS

POWER CONDITIONING
8.2 KW X 90 LBS/KW
= 740 LBS

FIGURES OF MERIT
(W/LB)

DAYTIME:	3.4
NIGHTTIME:	1.7
SYSTEM:	1.5 (W/FUEL) 2.5 (W/O FUEL)

Figure 18

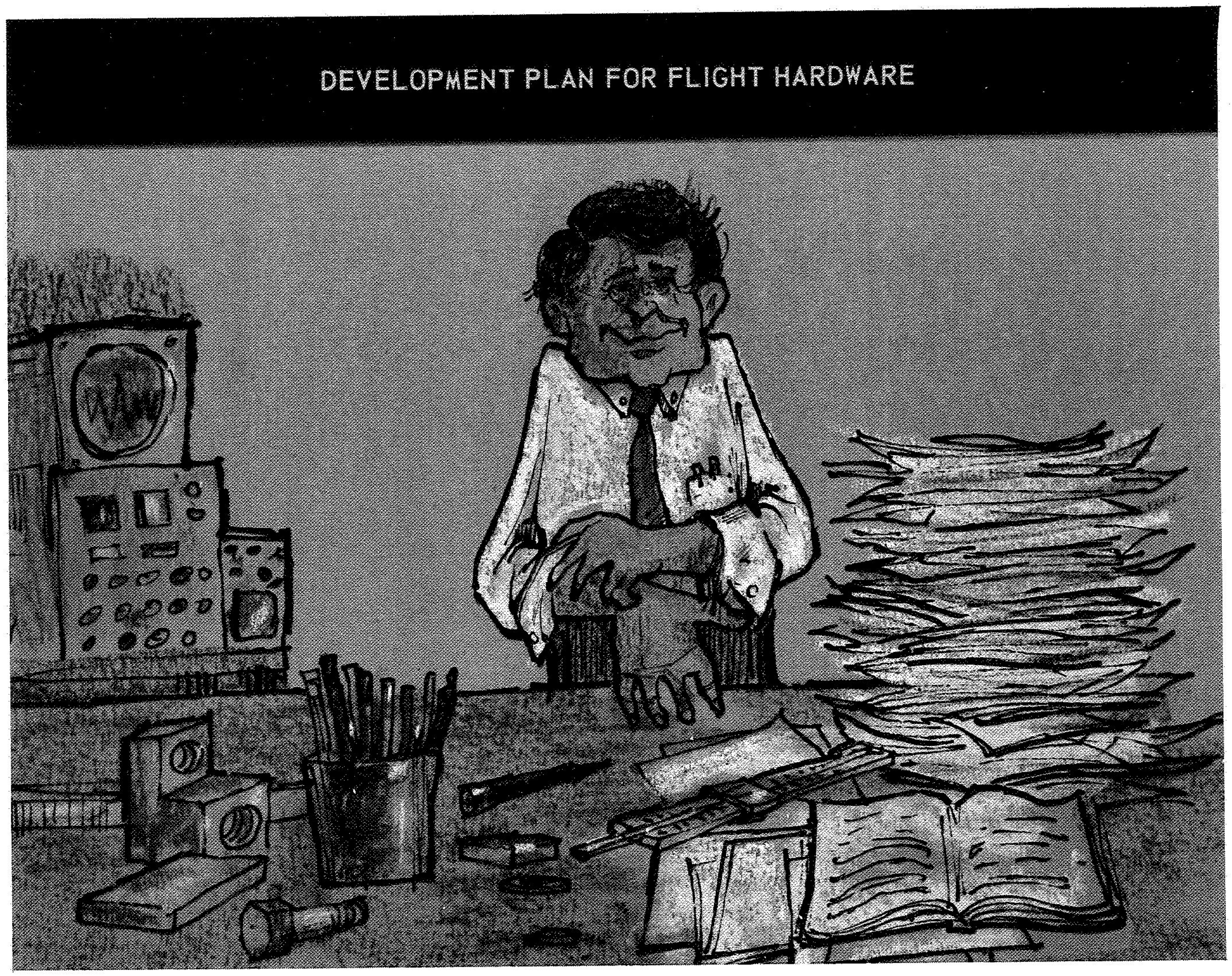


Figure 19

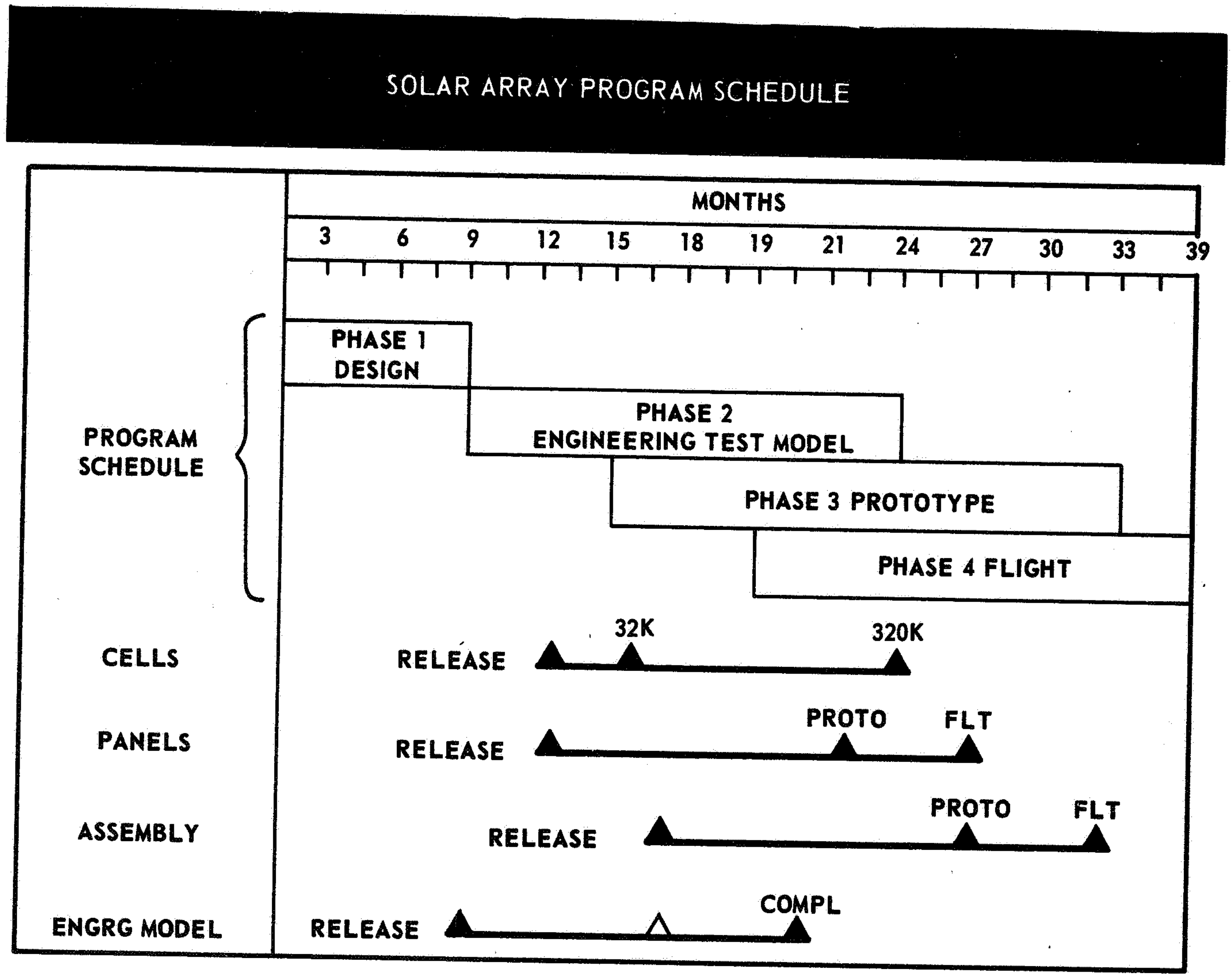

SOLAR ARRAY PROGRAM SCHEDULE
MONTHS
3 6 9 12 15 18 19 21 24 27 30 33 39
PROGRAM SCHEDULE
PHASE 1 DESIGN
PHASE 2 ENGINEERING TEST MODEL
PHASE 3 PROTOTYPE
PHASE 4 FLIGHT
CELLS
RELEASE
32K
320K
PANELS
RELEASE
PROTO
FLT
ASSEMBLY
RELEASE
PROTO
FLT
ENGRG MODEL
RELEASE
COMPL

Figure 20

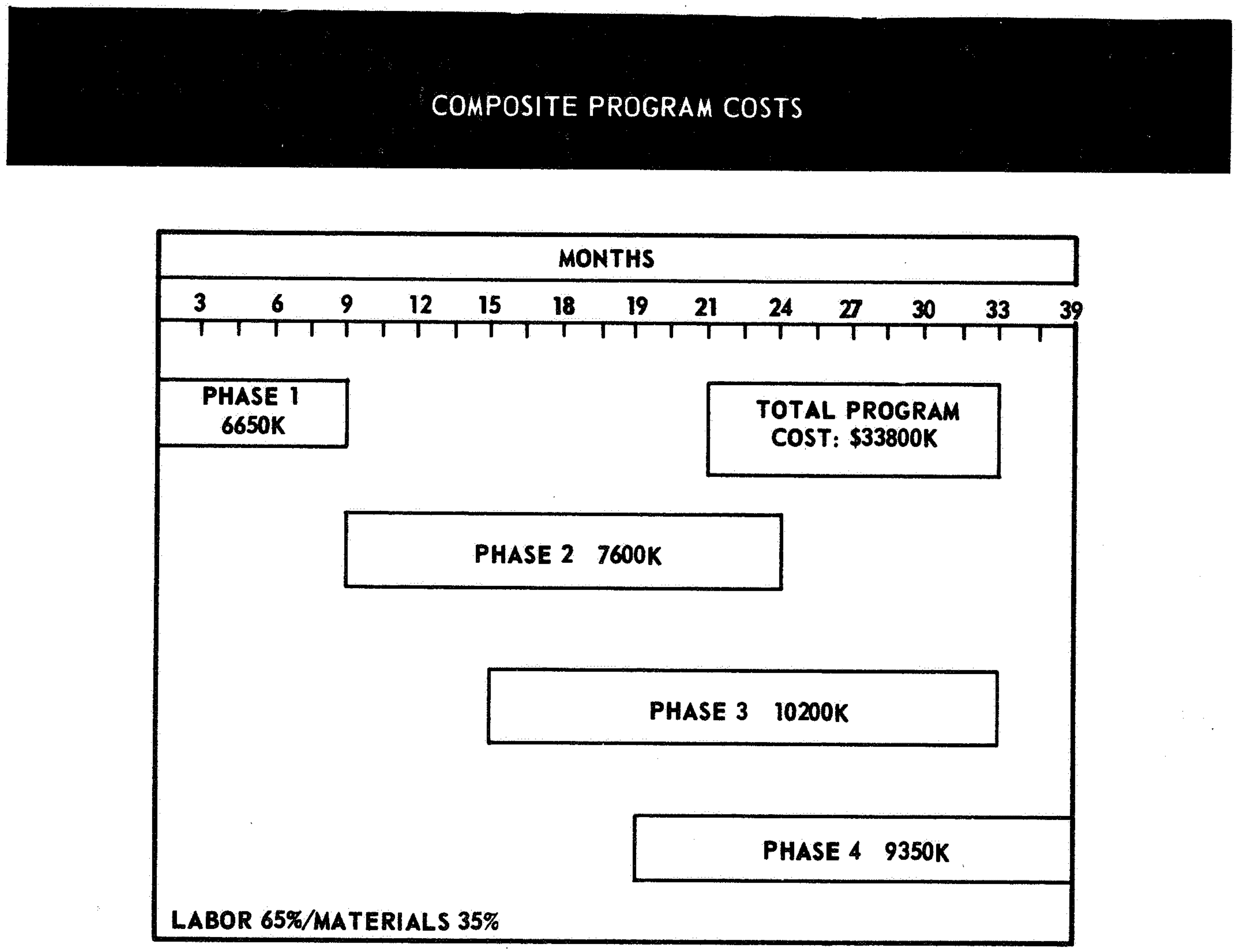

COMPOSITE PROGRAM COSTS
MONTHS
3
6
9
12
15
18
19
21
24
27
30
33
39
PHASE 1
6650K
TOTAL PROGRAM
COST: $33800K
PHASE 2 7600K
PHASE 3 10200K
PHASE 4 9350K
LABOR 65%/MATERIALS 35%

Figure 21

THE MAJOR SYSTEM ELEMENTS

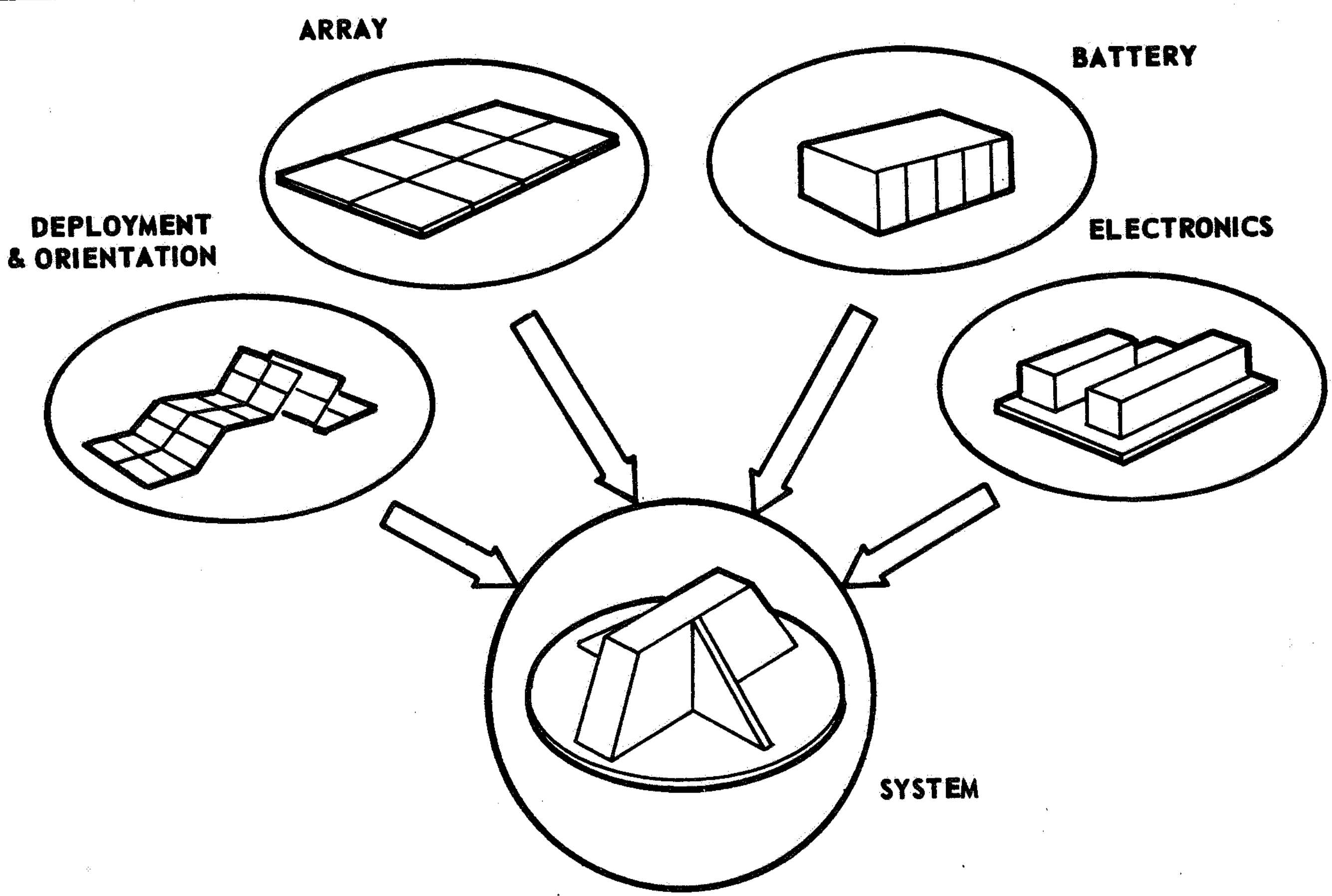


Figure 22

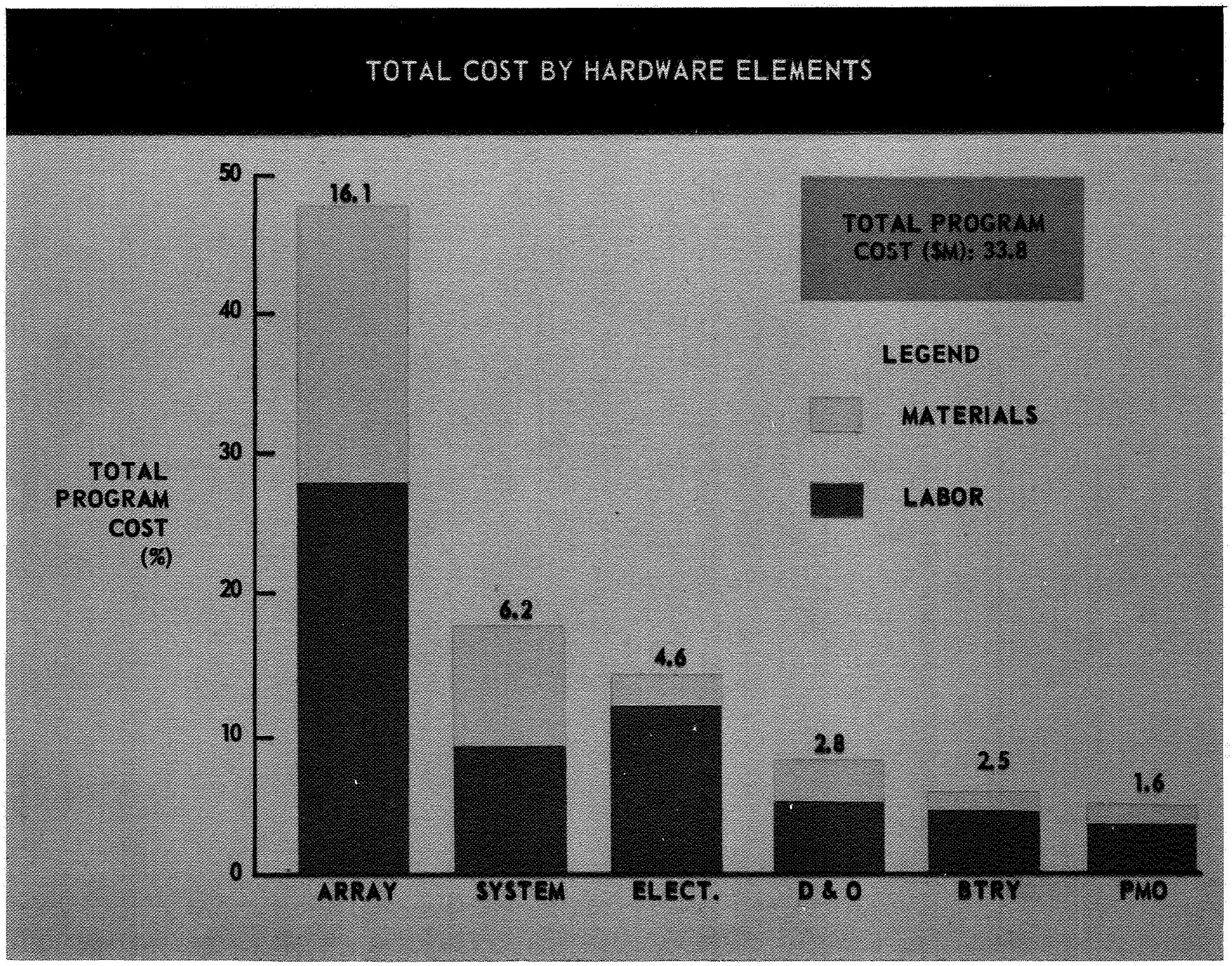

TOTAL COST BY HARDWARE ELEMENTS
TOTAL PROGRAM COST (%)
50
40
30
20
10
0
16.1
6.2
4.6
2.8
2.5
1.6
ARRAY
SYSTEM
ELECT.
D & O
BTRY
PMO
TOTAL PROGRAM COST ($M): 33.8
LEGEND
MATERIALS
LABOR

Figure 23

DETAILS OF SOLAR ARRAY

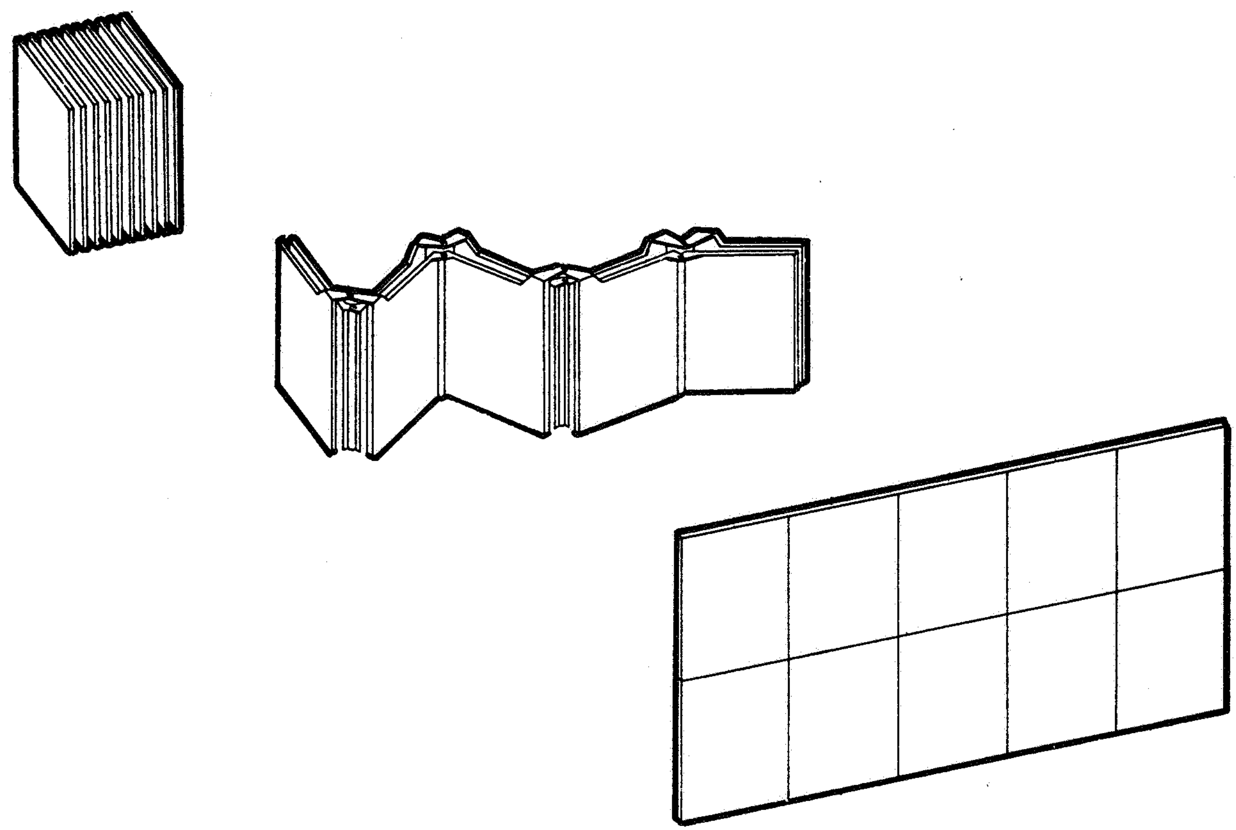


Figure 24

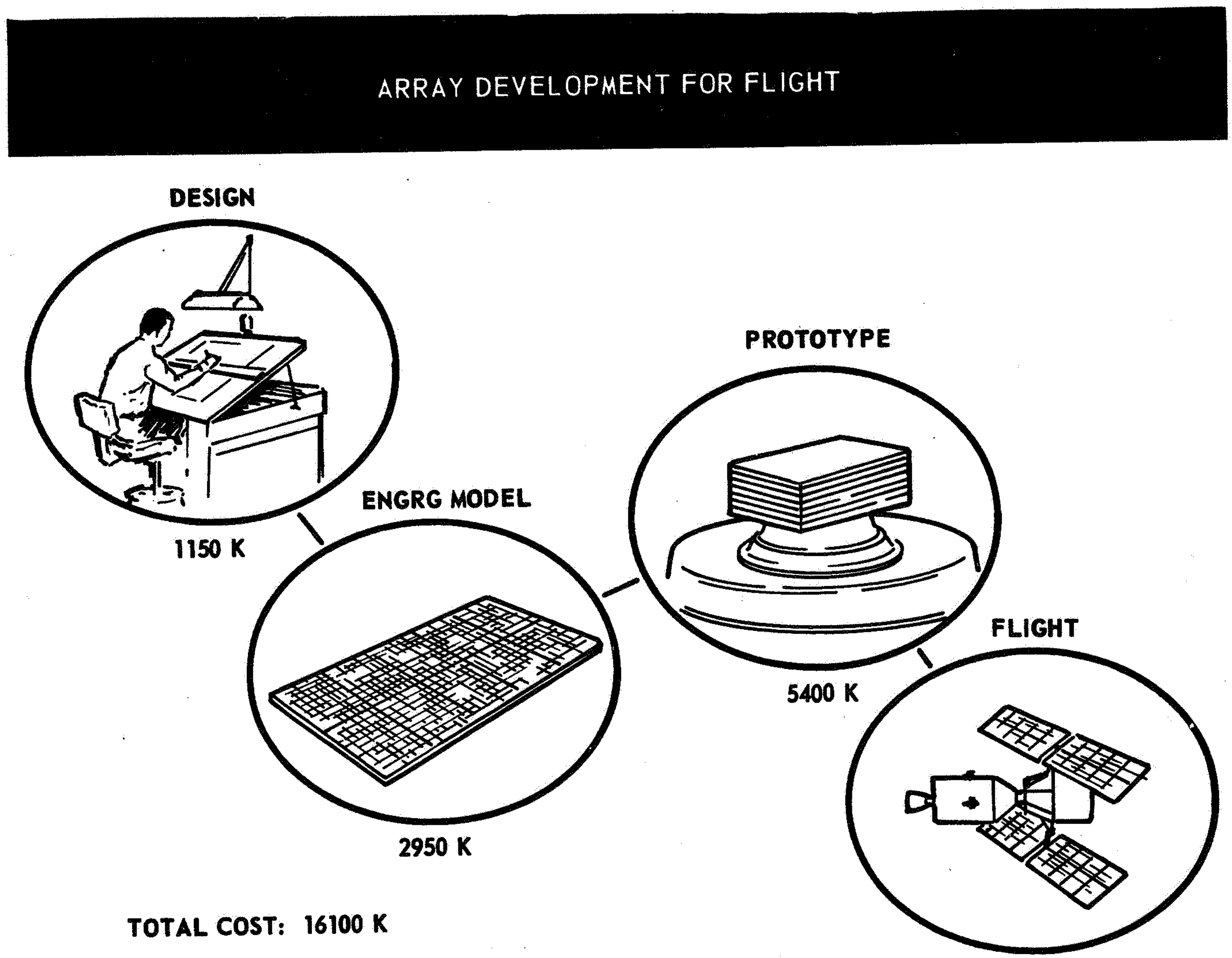

ARRAY DEVELOPMENT FOR FLIGHT
DESIGN
1150 K
ENGRG MODEL
2950 K
PROTOTYPE
5400 K
FLIGHT
TOTAL COST: 16100 K

Figure 25

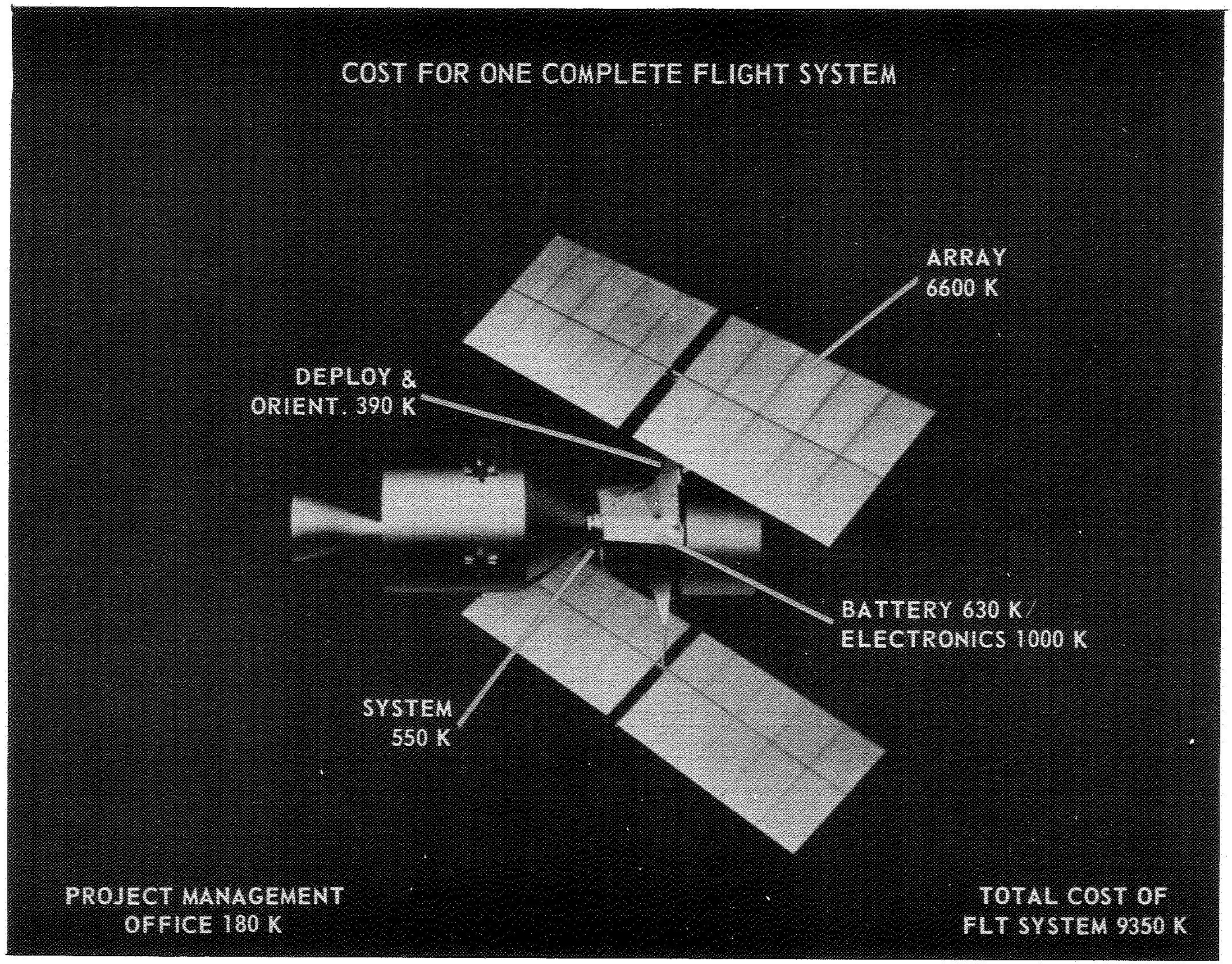

COST FOR ONE COMPLETE FLIGHT SYSTEM
ARRAY
6600 K
DEPLOY &
ORIENT. 390 K
BATTERY 630 K/
ELECTRONICS 1000 K
SYSTEM
550 K
PROJECT MANAGEMENT
OFFICE 180 K
TOTAL COST OF
FLT SYSTEM 9350 K

Figure 26

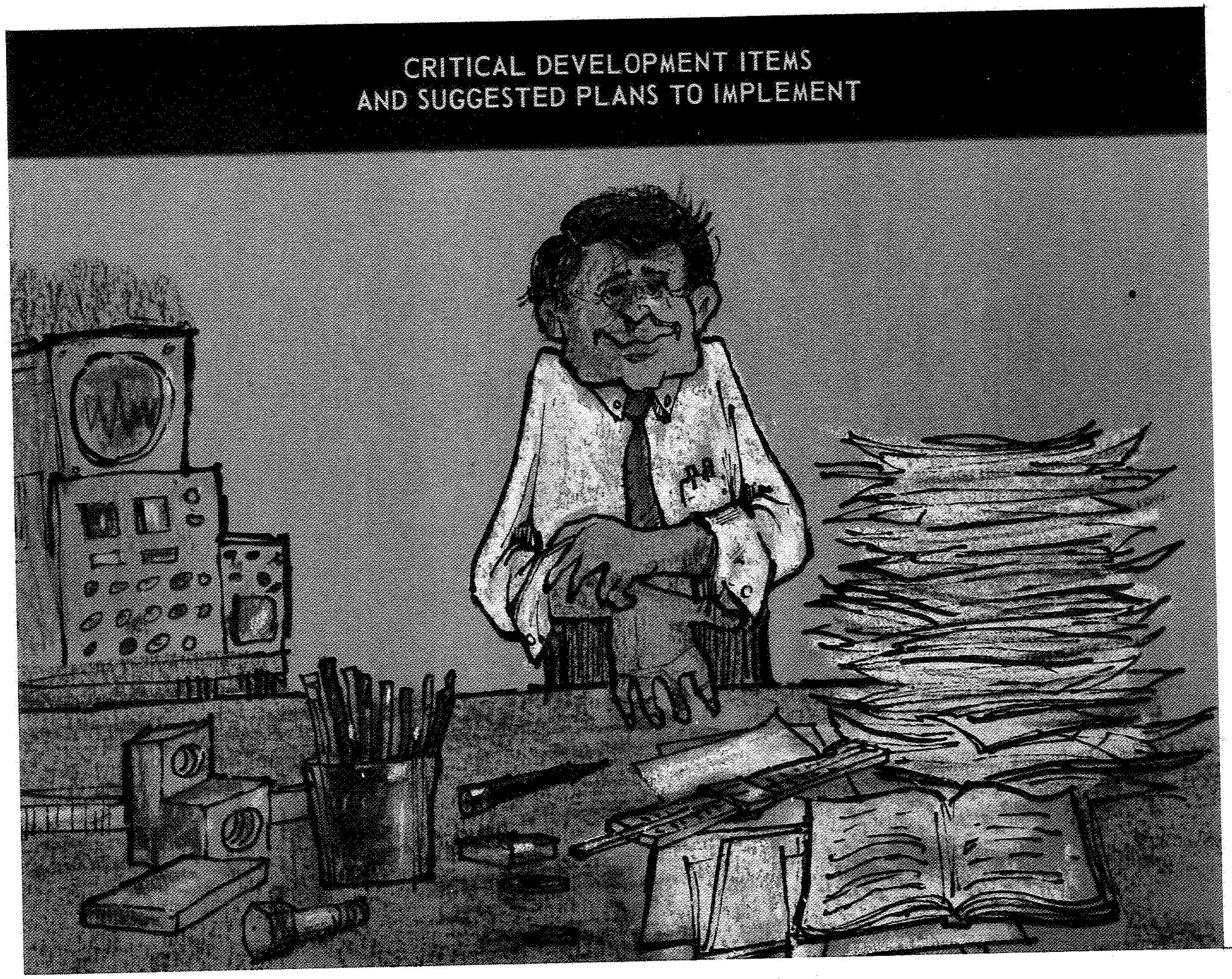

CRITICAL DEVELOPMENT ITEMS
AND SUGGESTED PLANS TO IMPLEMENT

Figure 27

DEVELOPMENT TASKS

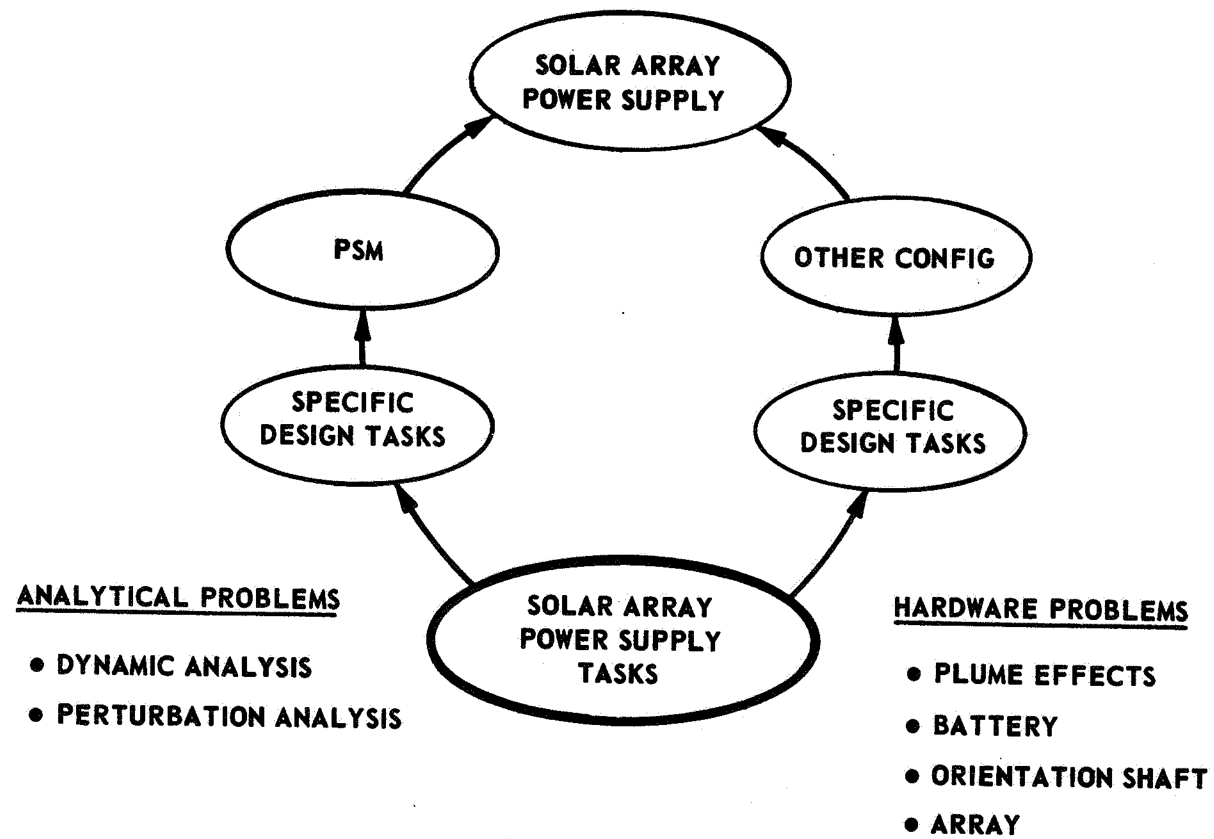


Figure 28

PERTURBATION FORCE SOURCES

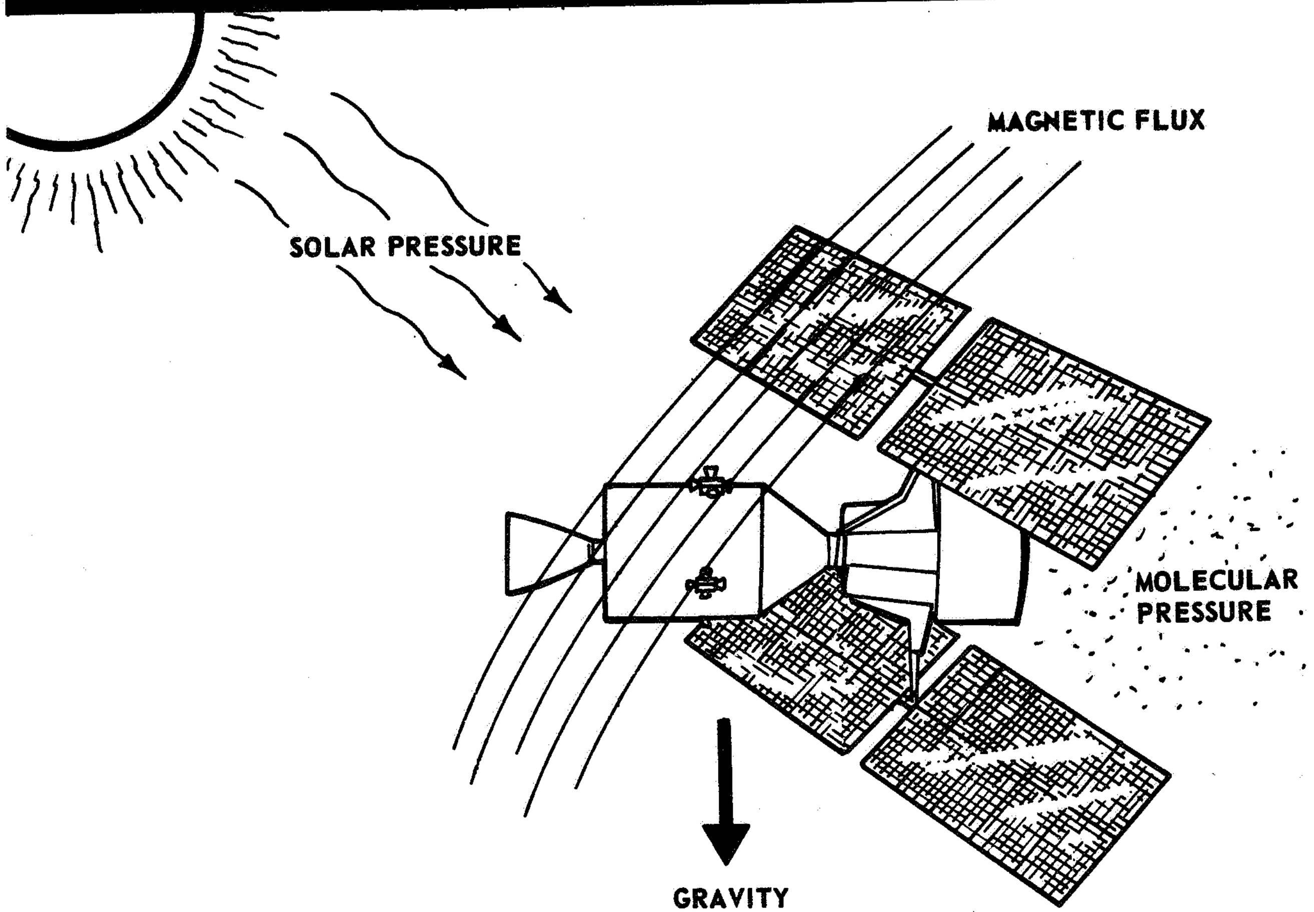


Figure 29

PERTURBATION EFFECTS ANALYSIS

- **ALTITUDE**
  - — DRAG
  - — LIFT
  - — MAGNETIC DIPOLE
  - — GRAVITY GRADIENT

- **INCLINATION**
  - — MAGNETIC DIPOLE
  - — SUN VECTOR – DRAG, LIFT, GG
  - — SOLAR PRESSURE

Figure 30

PERTURBATION EFFECTS ANALYSIS (CONT'D)

- SUN ELEVATION
  - SUN VECTOR – DRAG, LIFT, GG
  - SOLAR PRESSURE
- ARRAY SIZE AND ASPECT RATIO
  - EFFECTS ALL VARIABLES
- ATTITUDE ACCURACY
  - FUEL CONSUMPTION

Figure 31

PERTURBATION PROGRAM PLAN

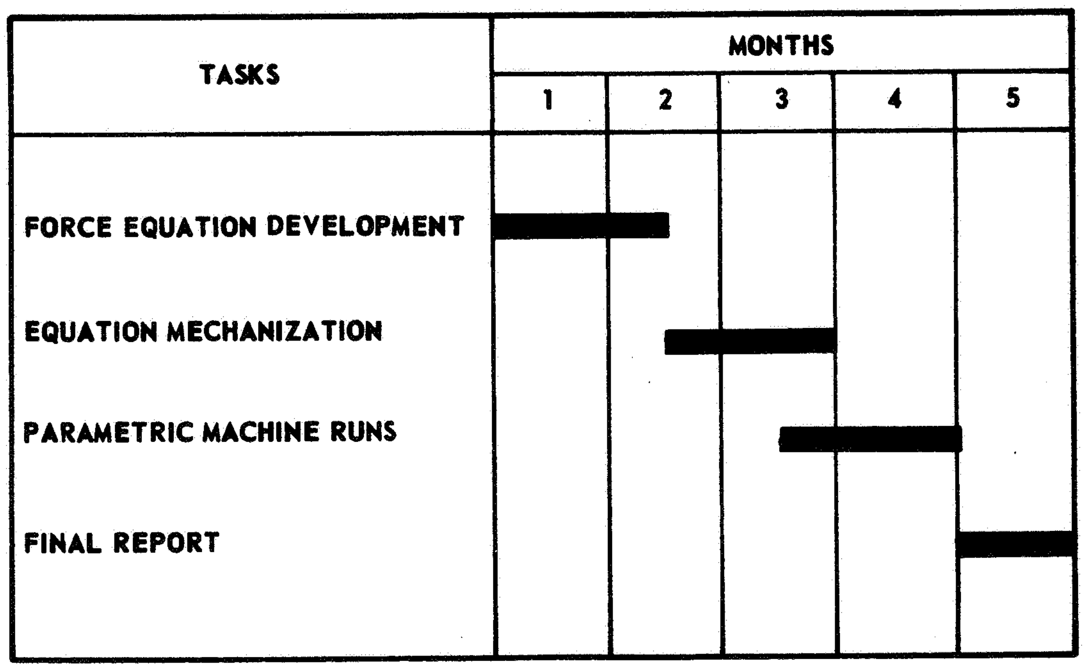


Figure 32

DYNAMIC PERTURBATION

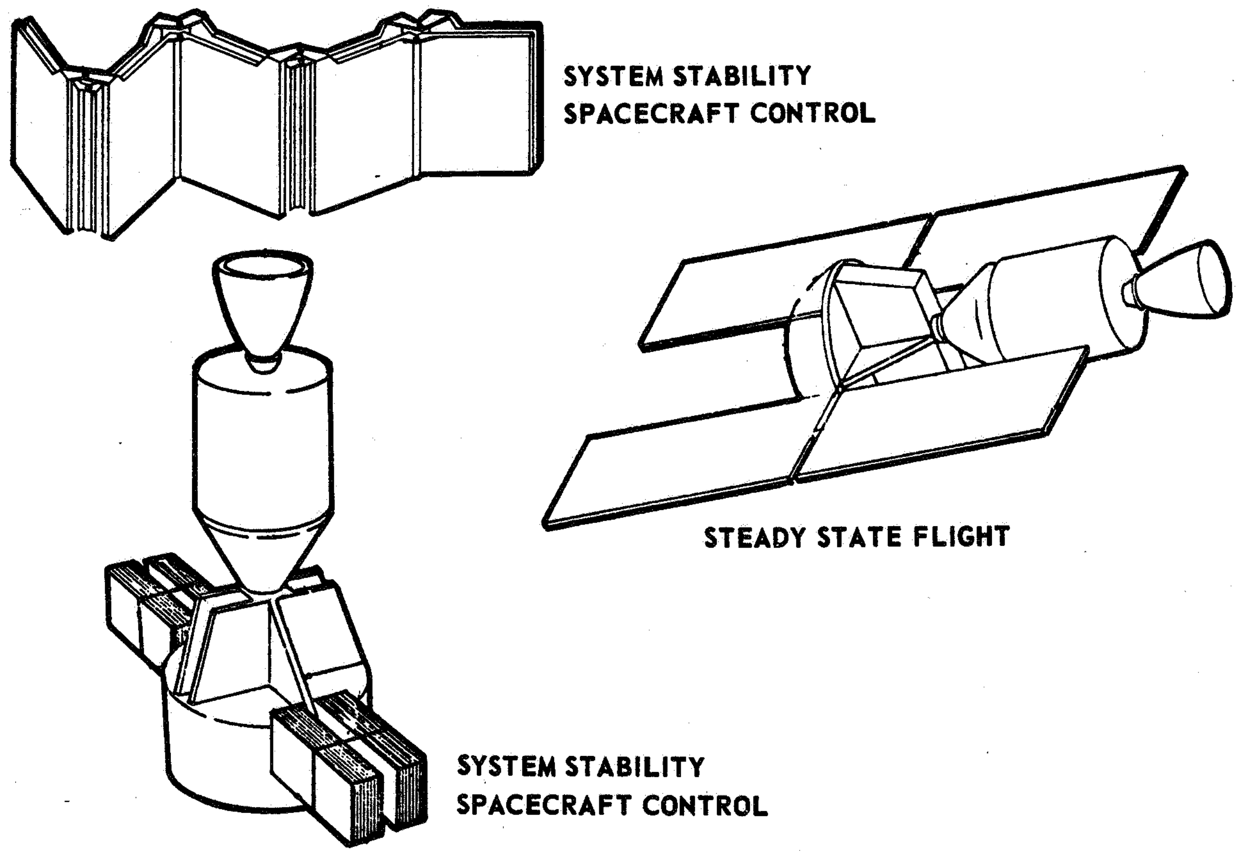


Figure 34

SPACECRAFT, ARRAY & SUN TRACKER AXIAL RELATIONSHIPS

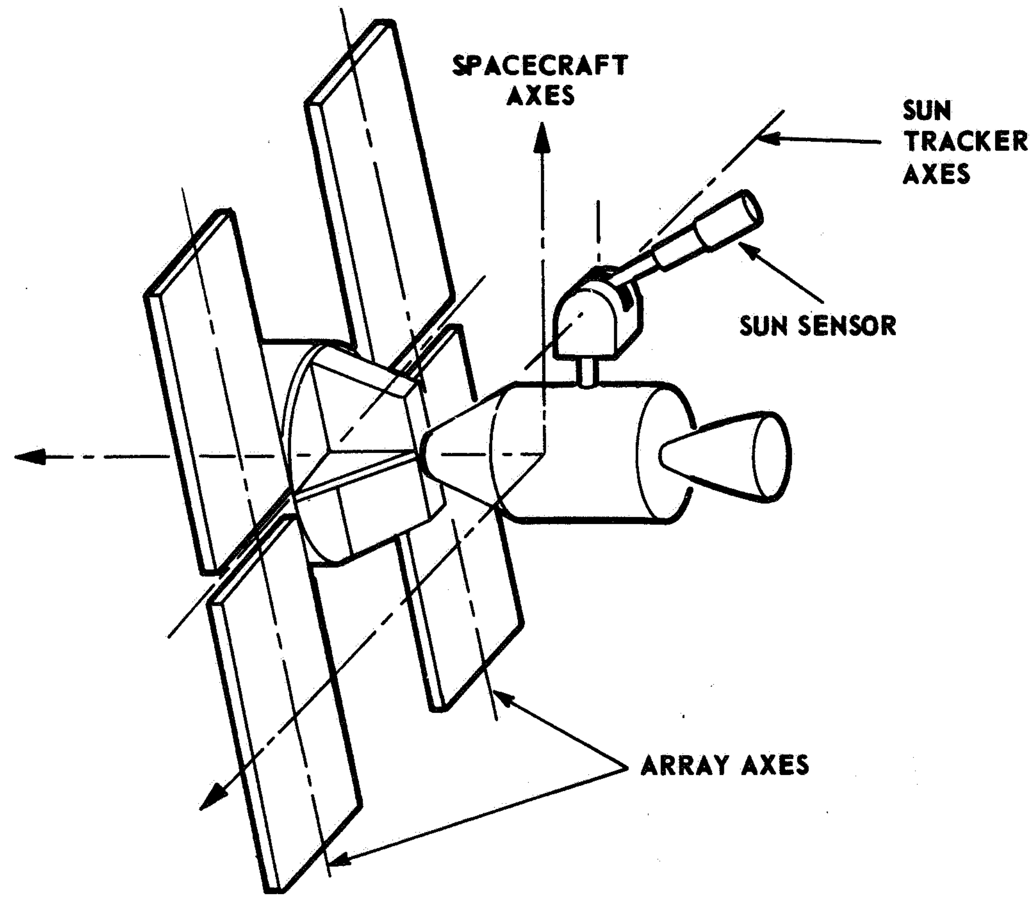


Figure 35

VEHICLE ARRAY INTERACTION DIAGRAM

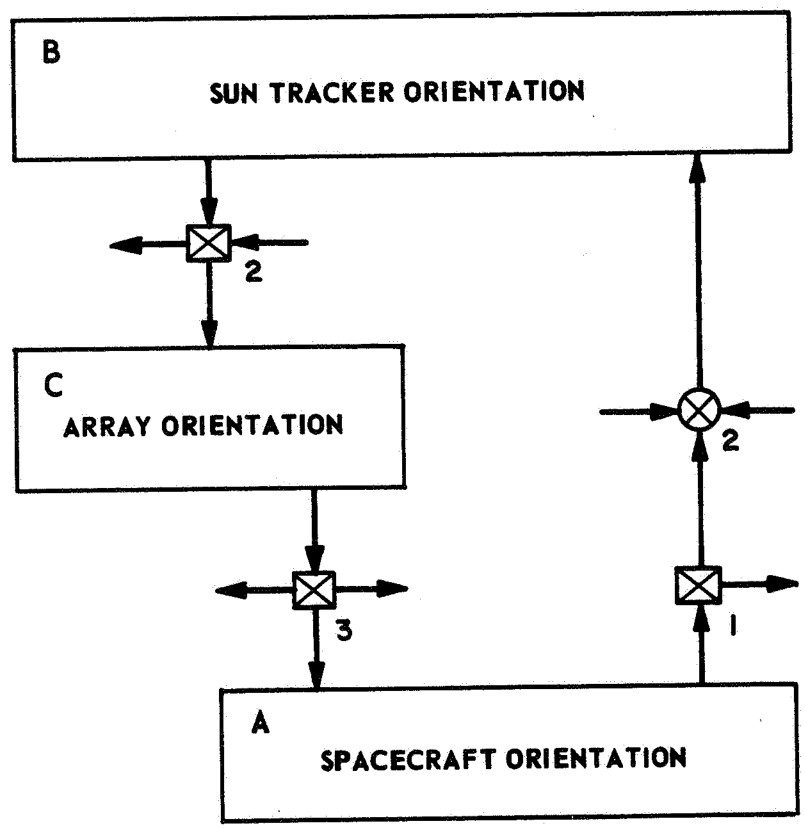


Figure 36

DYNAMIC ANALYSIS PROGRAM PLAN

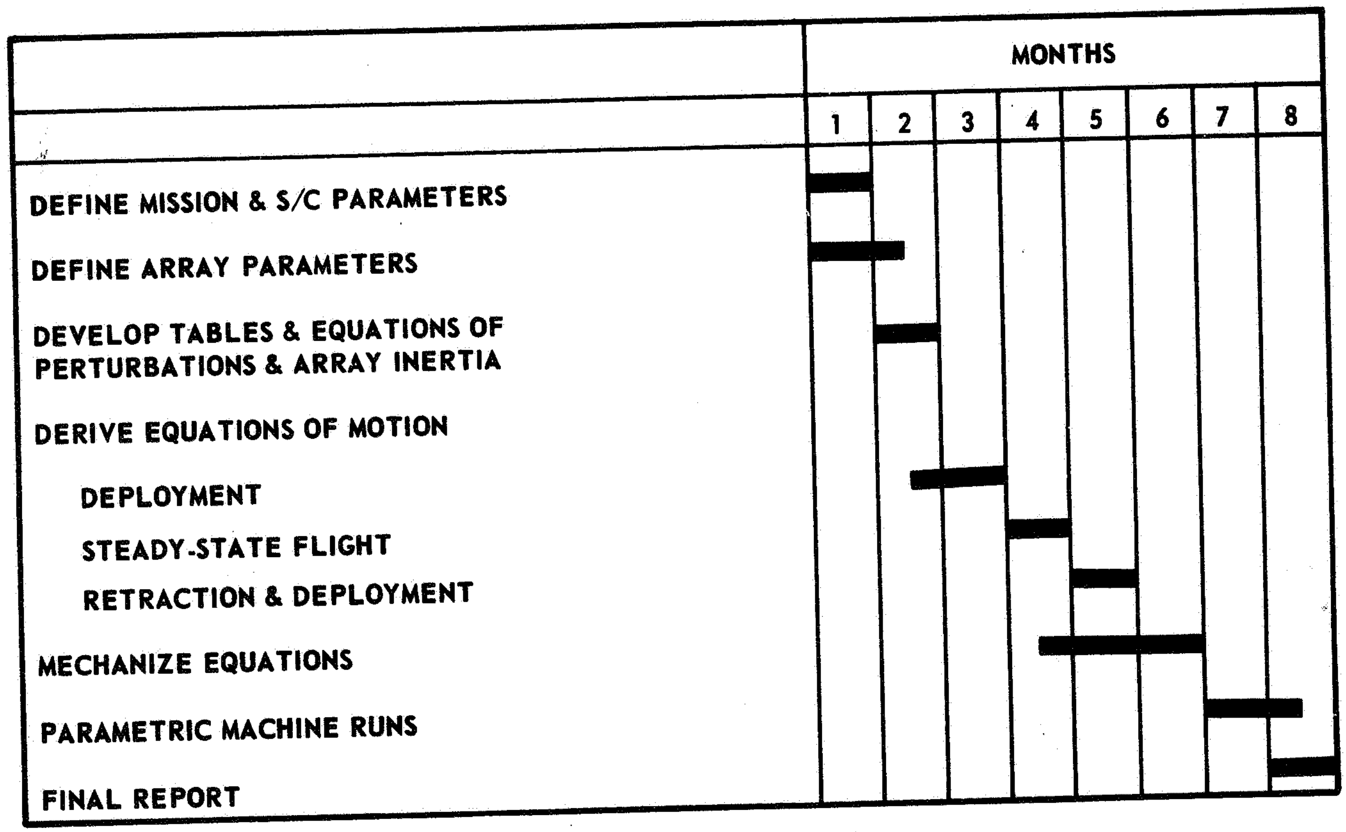


Figure 37

PROGRAM YIELD

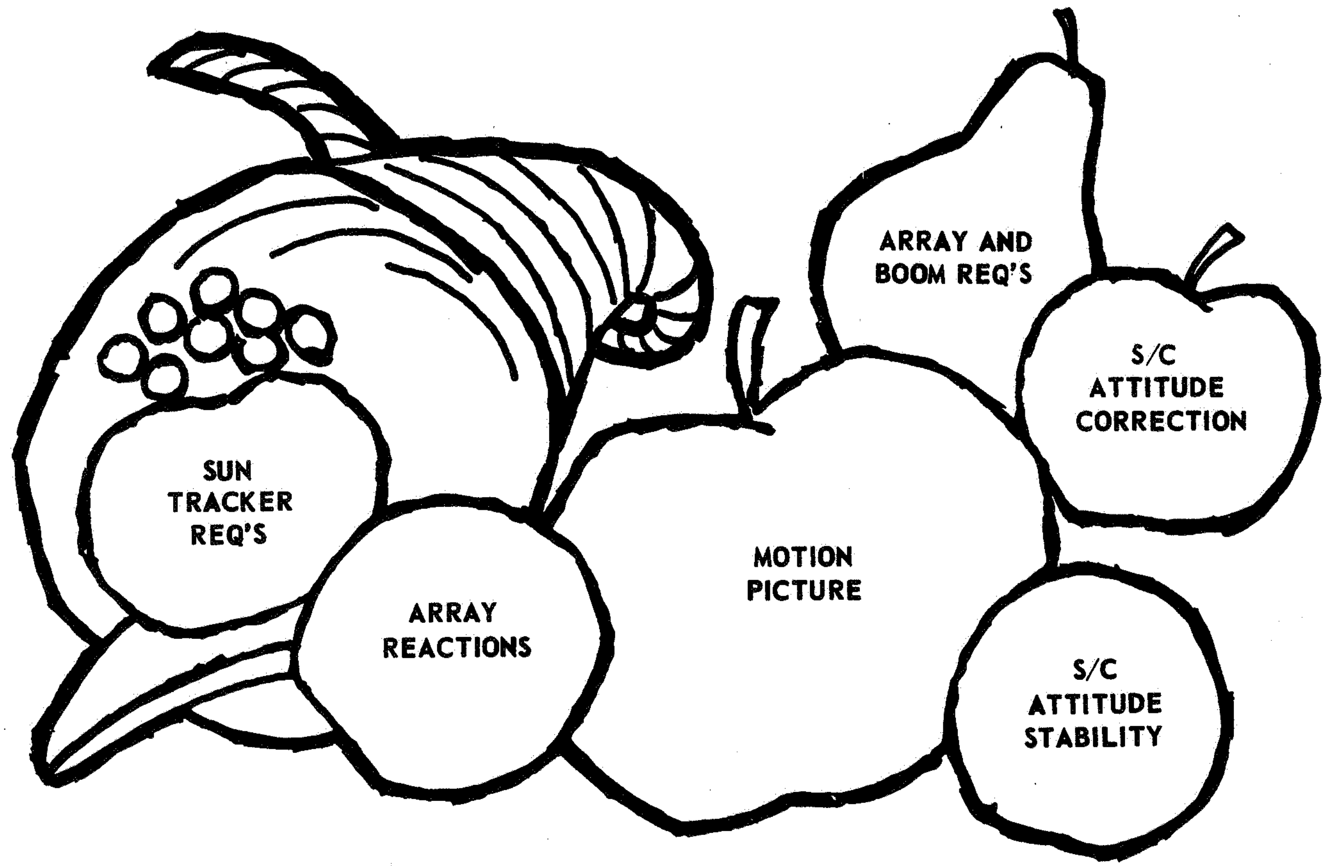


Figure 38

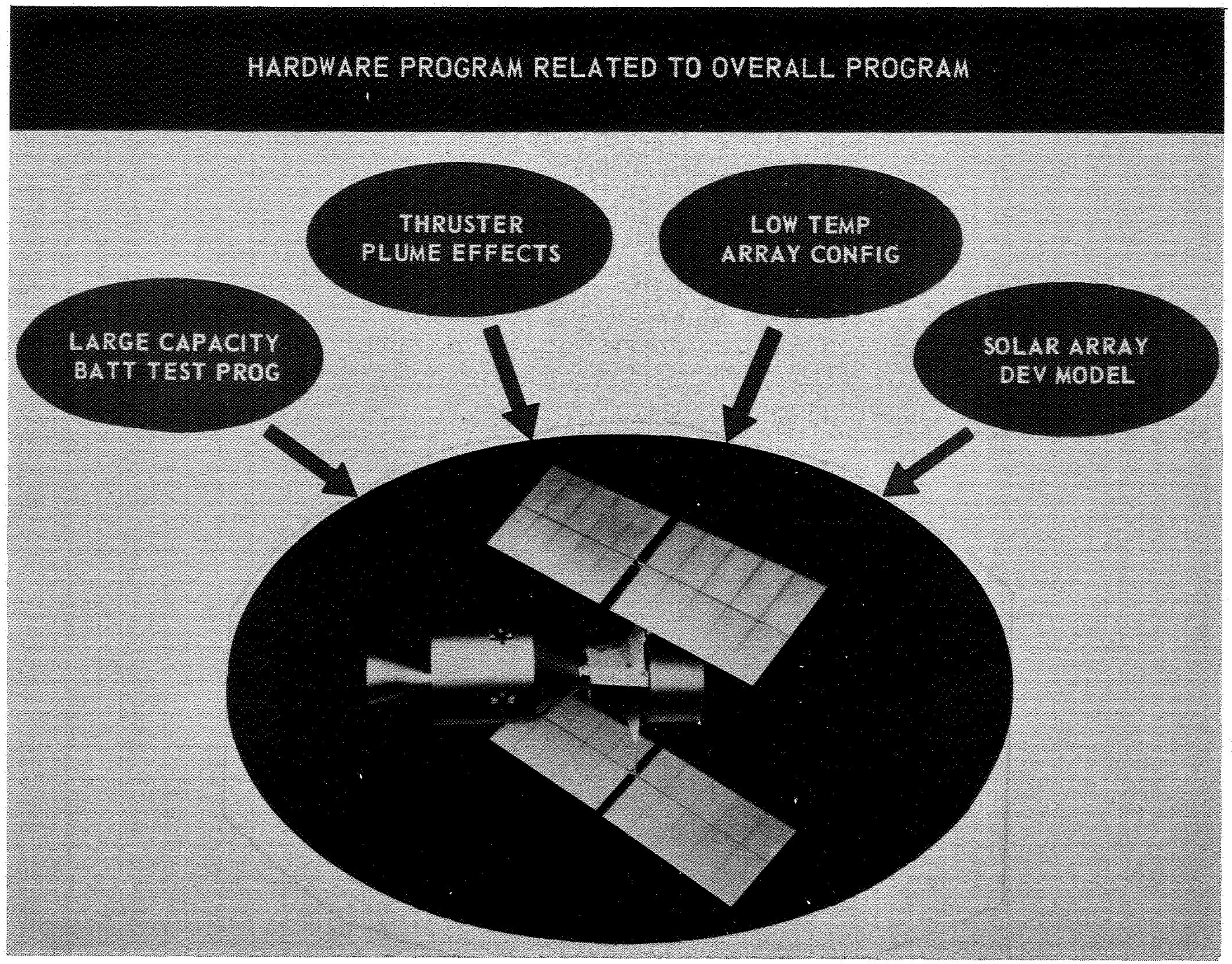

HARDWARE PROGRAM RELATED TO OVERALL PROGRAM
LARGE CAPACITY BATT TEST PROG
THRUSTER PLUME EFFECTS
LOW TEMP ARRAY CONFIG
SOLAR ARRAY DEV MODEL

Figure 39

MAJOR ELEMENTS OF ARRAY

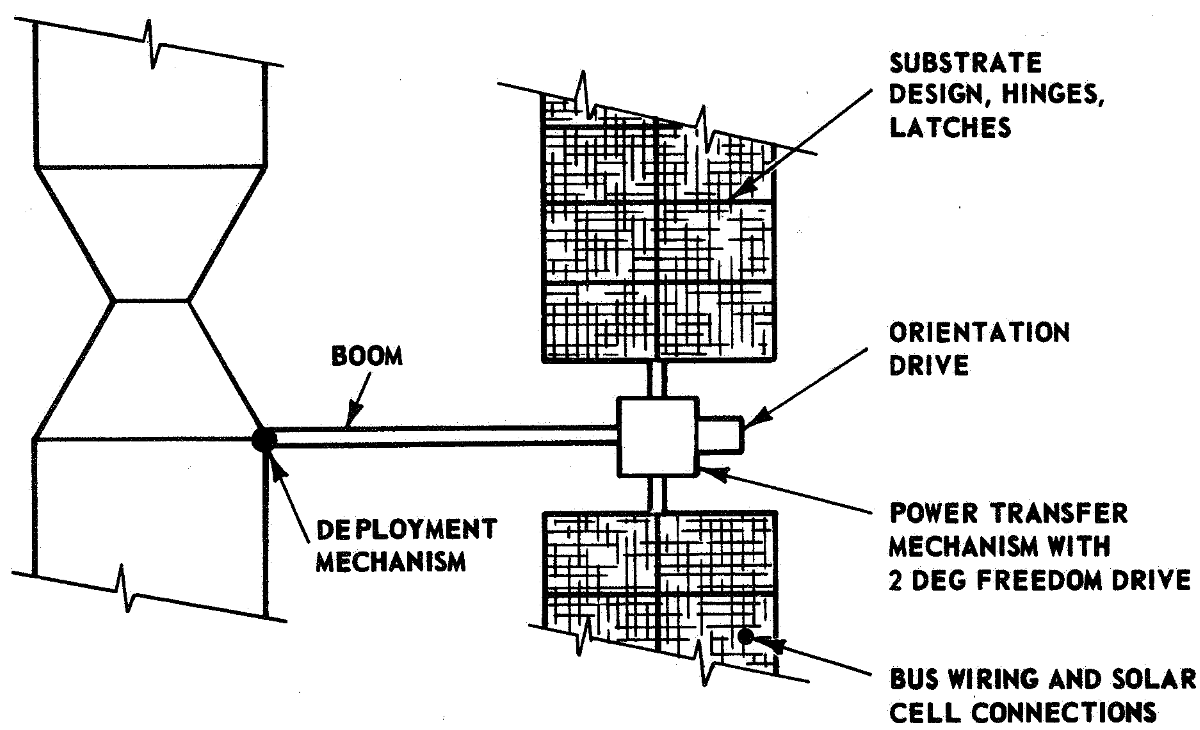


Figure 40

SOLAR ARRAY DEVELOPMENT MODEL

PURPOSE

- DEMONSTRATE DESIGN OF DEPLOYMENT, AND
- RETROACTION TECHNIQUES AND MECHANISMS

PLAN

- CONFIGURATION AND TEST DEFINITION
- DESIGN ARRAY SECTION
  - HINGES
  - LATCHES
  - DRIVE SYSTEM
  - DRIVE SHAFT (BOOM)
  - DEPLOYMENT MECHANISM

Figure 41

SOLAR ARRAY DEVELOPMENT MODEL (CONT'D)

PLAN

- DESIGN AND TEST MINOR TEST ARTICLES
- FABRICATE MAJOR TEST ARTICLES
- TEST MAJOR ARTICLE
  - — VIBRATION (STOWED)
  - — VIBRATION AND SHOCK (DEPLOYED)
  - — VIBRATION AND SHOCK (RETRACTED)
  - — THERMAL VACUUM
  - — MECHANISM LIFE TESTS
- PREPARE SPECIFICATION

TIME OF PROGRAM

- 18 MONTHS INCLUDES MECH LIFE TESTS

Figure 42

A P P E N D I X

RCA PRESENTATION TO NASAS AND INDUSTRY

July 13, 1967

Attendance List

NAME	ORGANIZATION
R.V. Hensley	Program Analysis Branch, OART, NASA
M. Sandel	Asst. Chief, Publications Branch NASA
William H. Woodward	Director, Space Power & Electric Propulsion OART, NASA
S. V. Manson	Head, Advanced Component Technology, OART, NASA
B. L. Dorman	Assistant Administrator for Industry Affairs, NASA
D. Novick	Chief, Space Flight Experiment Branch, OART, NASA
V. N. Huff	Systems Engineering, Manned Space Flight, NASA
A. H. Smith	Chief, Solar and Chemical Power Systems,OART, NASA
P. T. Maxwell	Head, Distribution Control Technology, OART,NASA
E. M. Cohn	Head, Electrochemical Systems, OART, NASA
N. J. Mayer	Space Vehicle Structure Program, OART, NASA
J. P. Mullin	Head, Electrothermal & Electromagnetic, OART, NASA
C. P. Mook	Environmental Factors & Aerodynamics Program OART, NASA
J. R. Dawson	Space Structures Branch, Langley Research Center
J. Patterson	Spacecraft Power Systems Sec., Langley Research Center
A. Obenschain	Space Power Tech. Branch, Goddard Space Flight Ctr.
Robert Loucks	Spacecraft Power Section, JPL
R. P. Arno	MAD, Ames Research Center
M. Baltas	Communications Program, Office of Space Science and Applications, NASA
J. Lehmann	Mgr. Advanced Programs & Technology Prog. Office of Space Science and Applications, NASA

NAME	ORGANIZATION
Joseph Haynos	FHC
Allen Osborne	Ryan Aeronautics Company
J. Lafleur	Atomic Energy Commission
William W. Hough	BELLCOMM MLS
B. W. Moss	BELLCOMM
L. A. Ferrara	BELLCOMM
H. Elakman	Electro-Optical Systems
Irving Stein	RCA
Herbert Bilsley	RCA
R. A. Newell	RCA
George Wolff	Hughes Aircraft
B. Hoonidge	Hughes Aircraft
Harry Adamitz	Boeing
James Burnett	Electro-Optical Systems
Clifford H. Paul	RCA/AED
George Barna	RCA/AED
Dan Mager	RCA/AED
Guy Yllmo	RCA/Washington
Bernard Mirosky	General Electric, Valley Forge
B. Crowe	CSI, Falls Church, Va.
C. E. Johnson	BELLCOMM
D. Macchia	BELLCOMM
L. Ule	North American Aviation